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MANNED SPACECRAFT CENTER
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SERVICE PROPULSION SYSTEM

By Joseph Fries
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SUMMARY

The performance of the Spacecraft 020 service propulsion system during the Apollo 6 mission was analyzed using the Apollo propulsion analysis program. A satisfactory correlation of the flight data was achieved. From the analysis, it was determined that the flight performance was as follows before crossover: thrust, 20,840 pounds; specific impulse, 310.2 seconds; and mixture ratio, 1.984. After crossover, the flight performance was determined to be the following: thrust, 21,360 pounds; specific impulse, 310.3; and mixture ratio, 2.0. The engine flight performance corrected to standard inlet conditions resulted in the following values: thrust, 21,357 pounds; specific impulse, 309.8 seconds; and mixture ratio, 2.014. These values are 0.31, 0.03, and 0.20 percent higher, respectively, than the standard inlet condition values reported in the acceptance test log. These differences are within the accuracy with which these values can be determined during engine acceptance testing.

The Apollo 6 service propulsion system operation was nominal, and the two detailed test objectives — no-ullage start and long-duration burn — were accomplished to a satisfactory degree. The only apparent hardware problem that was encountered on the Apollo 6 service propulsion system was the chamber pressure overshoot which had no adverse effect on the propulsion system.

Several recommendations are being made to eliminate problems encountered during the analysis of the Apollo 6 mission. These recommendations are as follows: The Block II propellant loading procedures should require more than one propellant sample be taken, and the density of these samples should be measured and reported to the NASA MSC. An investigation should be made to assure that the Block II propellant loading procedures properly describe the gaging system calibration procedures. A procedure should be established for the calculation and reporting of the propellant loads. The proper gaging system calibration

curves should be supplied to the NASA MSC. All calibration curves should be for the flight telemetered transducers and not class calibrations.

INTRODUCTION

Analysis Description

The Apollo 6 mission was another in a series of unmanned flight tests aimed at the qualification of the launch vehicle and spacecraft systems. This report is devoted entirely to the propulsion performance analysis of the Apollo 6 service propulsion system (SPS) and to presentation of the analysis results.

The primary objective of the analysis is to determine the steady-state performance of the SPS under the environmental conditions encountered in space. The approach employed to satisfy this objective is one of a minimum-variance estimation technique. This technique utilizes all pertinent flight and ground test data and statistically evaluates the data to best reconstruct the acceleration data derived from the guidance computer output.

In order to accurately determine the steady-state propulsion performance, the data used have to be evaluated to eliminate sources of error. The data evaluation also serves as a critique of the major systems of the SPS and the servicing of these systems. Thus, an analysis of the gaging system and the pressurization system is presented. Associated with the gaging system, but presented as a separate topic, is the propellant-loading procedure and analysis.

An evaluation of the engine start and shutdown transients is performed and compared with previous flights. The chamber pressure overshoot experienced during start is also discussed in this section.

The detailed test objectives for the Apollo 6 SPS are also evaluated. The flight data and performance analysis results are compared to the individual success criteria for each objective and deviations are discussed.

The conclusion is intended to be a critique of the overall propulsion system operation and to point out problem areas present. The problem areas are not necessarily associated with the propulsion system hardware but also deal with mission planning and prelaunch operations.

Service Propulsion System Mission Description

The Apollo 6 spacecraft (SC 020) was placed into orbit by the Saturn V launch vehicle Apollo-Saturn 502 (AS-502). Launch occurred at approximately 7:00 a.m. (e.s.t.) on April 4, 1968, from pad 39, Kennedy Space Center (KSC). The Apollo 6 mission included the fourth flight test of the SPS. The primary SPS test objectives were to demonstrate a satisfactory start without a reaction control system (RCS) propellant settling maneuver and to determine the SPS performance during a long-duration burn. The Apollo 6 mission plan called for two SPS burns; the first burn being approximately 254 seconds in duration and a subsequent burn of approximately 189 seconds.

Because of an inability to restart the Saturn IVB (S-IVB) stage of the launch vehicle, an alternate mission plan was implemented. The alternate plan called for a single SPS burn to provide the ΔV required to transfer from its earth parking orbit to a highly elliptical earth intercepting orbit to satisfy the reentry condition of the heat shield test. Upon a guidance and navigation initiated command, the SPS ignited at 3:16:06.20 ground elapsed time (g.e.t.). Shutdown occurred at 3:23:27.91 g.e.t., resulting in a burn duration of 441.71 seconds. The firing was not preceded by an ullage maneuver.

Subsystem Description

The SPS consists of three primary subassemblies: (1) engine system, (2) propellant storage and feed system, and (3) pressurization system. A functional flow diagram is shown in figure 1.

The Block I engine system produces a nominal thrust of 21 500 pounds, operating at nominal mixture ratio of 2.0. The combustion chamber is ablatively cooled. The propellants are earth storable and hypergolic. The fuel, Aerozine 50 (A-50), is a 50/50 blend (by weight) of unsymmetrical dimethylhydrazine and anhydrous hydrazine; the oxidizer is nitrogen tetroxide. The engine bipropellant shutoff valve is pneumatically actuated by gaseous nitrogen. Bolted to the engine chamber nozzle extension which consists of two columbium sections extending to an area ratio of 40:1 and a titanium section to the exit (62.5:1). The mode of nozzle extension cooling is by radiation to space.

Fuel and oxidizer are each contained in a set of two cylindrical tanks connected in series. The downstream tanks are called the sump tanks and are directly connected to upstream storage tanks by crossover lines and standpipes. Each sump tank contains propellant retention screens and a propellant reservoir which retain propellant over the propellant tank outlet during near zero-g conditions and reduce the propellant settling time requirements.

Thrust from the service module RCS engines provides for propellant settling in addition to that maintained by the above mentioned retention devices.

The helium pressurization supply is contained in two spherical pressure vessels at a nominal initial pressure of 4000 psia and ambient temperature and is isolated from the fuel and oxidizer tanks during engine shutdown by two normally closed solenoid valves. Two dual-stage regulators, arranged in parallel, are located downstream of the solenoid valves and provide pressure-regulated helium to the fuel and oxidizer tanks. Two sets of check valve assemblies, arranged in series-parallel configurations, prevent fuel or oxidizer from entering the pressurization system. Pressure relief valves prevent overpressure in the propellant tanks. Heat exchangers are used in the helium lines to condition the helium to a temperature approximately that of the propellant in the tanks.

The SPS hardware on Apollo 6 was identical to the SPS hardware on Apollo 4. The SPS propellant mass, however, was increased over the Apollo 4 loading to meet a greater impulse requirement.

PROPELLANT LOADING ANALYSIS

The SPS oxidizer tanks were loaded with nitrogen tetroxide on March 18, 1968. The oxidizer sump tank was filled to the top of the crossover line standpipe, and the sump tank primary gaging system output was adjusted to this known quantity. The oxidizer storage tank was then loaded by overflowing the sump tank through the crossover line. The filling flow rate was held at approximately 15 gallons per minute to minimize the entrainment of helium from the sump tank ullage, and thereby reduce the sump tank overfill that was observed during the loading of the Apollo 4 spacecraft. The storage tank was filled to the top point sensor, and the storage tank primary gaging system output was adjusted to this known quantity. The storage tank was then drained, through the sump tank, to the flight load, as indicated on the storage tank primary gaging system probe. At this point the storage tank gaging probe indicated 6820 pounds, the sump tank probe indicated 14 820 pounds, and the storage tank ullage pressure was 113 psia. The indicated sump tank overfill (above top of standpipe) was approximately 80 pounds. When the storage tank ullage was test pressurized to 175 psia, the storage tank probe indicated 6520 pounds and the sump tank probe indicated its maximum reading of 15 000 pounds.

The fuel tanks were loaded with A-50 on March 19 in a manner similar to the oxidizer loading. With the flight load on board and a storage tank ullage pressure of 97 psia, the storage tank primary gaging

system probe indicated 3320 pounds and the sump tank probe showed 7410 pounds. There was an indicated sump tank overfill of approximately 50 pounds. When the storage tank ullage pressure was raised to 175 psia, the storage tank probe indicated 3140 pounds and the sump tank probe indicated its maximum reading of 7500 pounds.

A density determination was made for one oxidizer sample (sample G35758) taken from the SPS oxidizer supply. The sample density was reported to be 1.4831 g/ml at 4° C. This value compares very closely to the mean value (1.4830 g/ml) of eight samples taken prior to the Apollo 4 mission. The sample density yielded a value of 90.25 lbm/ft^3 at the loaded oxidizer temperature of 70° F and the pressure of 113 psia.

A fuel density of 0.9032 g/ml at 25° C was reported for one fuel sample (sample G35962). This value results in a fuel density of 56.64 lbm/ft^3 at 70° F and a pressure of 97 psia. The reported Apollo 6 fuel density is approximately 0.3 percent higher than the fuel density reported for the Apollo 4 mission and is at the upper limit of the applicable military specification for A-50. It is possible that the reported fuel density is in error. The Bendix laboratory is investigating this possibility; however, to date, no other value of the fuel density has been obtained.

The reported propellant densities were used to develop the following equations for use during the Apollo 6 evaluation. The equations were based on the Titan Handbook equations with adjustments made to the intercepts to match the reported propellant densities.

$$\rho_\phi = 95.64 - 0.078035(T) + 0.000699(P - 14.7)$$

where ρ_ϕ = oxidizer density, lbm/ft^3

T = oxidizer temperature, °F

P = oxidizer static pressure, psia

$$\rho_f = 58.84 - 0.031838(T) + 0.000368(P - 14.7)$$

where ρ_f = fuel density, lbm/ft^3

T = fuel temperature, °F

P = fuel static pressure, psia

The propellant densities based on the reported sample densities and the gaging system readings during loading were used to compute the SPS propellant loads presented below.

Propellant	Total mass loaded, lbm		
	Actual	KSC reported	Planned
Oxidizer	[a]22 185	22 015	21 980.2
Fuel	[a]11 038	10 964	10 940.5
Total	[a]33 223	32 979	32 920.7

[a]Includes gageable, ungageable, and estimated vapor on board.

The reported loads and the actual calculated loads differ due to differing treatment of propellant densities, propellant mass in the crossover lines, and propellant vapor. The reported loads were derived assuming nominal densities and partially full crossover lines. The reported loads also assumed that the total mass read from the propellant loading tables included a vapor allowance. The actual loads include a correction to measured densities and an allowance for full crossover lines. The allowance for full crossover lines was made since the sump tanks were filled above the standpipe crossover lines. Since the reported loads were taken directly from the loading tables, they assumed partially full crossover line standpipes. However, because of the sump overfill, the correction described above was necessary. The actual loads also include the estimated vapor in the tanks under the loading conditions.

Efforts are presently underway to establish procedures for future missions to assure that the reported loads will be correct.

DATA PROCESSING

Upon receipt of the Apollo 6 mission station data tapes, the Data Reduction Center processed the tapes and produced a phase I tape. The phase I tape was quantified and packed onto a Univac 1108 compatible binary tape formatted for the Propulsion and Power Division (appendix). Also stripped off the phase I tape was the guidance computer word which was put out on a separate tape in the standard Apollo guidance computer down-link list format. These tapes were supplied to the Primary Propulsion Branch by the Computation and Analysis Division.

The Univac 1108 binary tape was then processed through a decommutation program which produces raw data plots and smoothes the raw data. The raw data were smoothed using an orthogonal polynomial sliding arc filter, the spans of which are presented in table I, and sliced at a sample rate of 1 sample per second for input to the analysis program. The raw data and smoothed data were plotted with Calcomp plotters.

The guidance computer data are also specially processed. The data, which were in the form of velocity increment counts, were first edited to eliminate bad data and then they were scaled, biased, smoothed (table I), sliced, and converted to acceleration. The acceleration data, which were also sliced at a sample rate of 1 sample per second, were merged with the smoothed propulsion system data. This resultant tape was the input tape to the analysis program.

STEADY-STATE PERFORMANCE ANALYSIS

Analysis Technique

The major effort for this report was concentrated on determining the SPS steady-state performance during the Apollo 6 mission. This was accomplished by utilizing the Apollo propulsion analysis program. The program utilizes a minimum variance estimation technique in conjunction with pertinent data from the flight and from previous static tests, in addition to the physical laws which describe the behavior of the propulsion/propellant systems and their interactions with the spacecraft. The program embodies error models for the various flight and static test data that are used as inputs, and by iterative methods arrives at estimations of the system performance history, initial propellant weights, and spacecraft weight which "best" (minimum-variance sense) reconcile

the available data. The technique is to determine the coefficients of the propulsion and propellant systems performance parameters in the error model that minimizes the quantity χ^2.

$$\chi^2 = \sum_{i=1}^{m} \sum_{j=1}^{n} \frac{\left(Z_{ij}^* - Z_{ij}\right)^2}{\sigma_{ij}^2}$$

where χ^2 - a function to be minimized

Z_{ij}^* = a measured data value

Z_{ij} = a point calculated by the simulation which corresponds to Z_{ij}^*

σ_{ij} = an a priori estimate of the standard deviation of the data point (which should include uncertainties both in the model and in the data)

m = the number of data measurements used

n = the number of data points per measurement

The key to a successful postflight analysis is the extremely accurate thrust acceleration that can be calculated from the Apollo guidance computer ΔV data. Assuming that there were no unknown biases present, it is estimated that the acceleration during the flight was determined within ±0.02 ft/sec^2. This would result in an accuracy of approximately 0.10 percent, which is more than an order of magnitude better than any other propulsion measurements. From the acceleration data, the time history of the ratio of thrust to weight can be determined. Fitting this ratio with the other sources of information previously mentioned and adjusting the initial conditions and measurements according to their estimated sigmas in an iterative procedure results in a converged condition which represents the best estimate of the true state.

Analysis Program Results and Critique of Analysis

The SPS steady-state performance was determined from the analysis of a 300-second segment of the SPS burn. The segment of the burn analyzed commenced approximately 39 seconds after SPS ignition (FS-1), and included the flight time between 11 805 and 12 105 seconds g.e.t.

The first 39 seconds of the burn were not included, to reduce any errors introduced by the non-steady-state conditions during start, or errors resulting from data filtering spans which include transient data. The segment was limited to 300 seconds because use of a longer segment would have required reliance on a predicted throat area time history which was increasingly questionable after approximately 350 seconds of total burn time. The 300-second segment analyzed contained burn times both before and after storage tank depletion (crossover), and is considered to be of more than sufficient duration to characterize the steady-state flight performance.

The Apollo propulsion analysis program results presented in this report were based on simulations using data from the flight measurements listed in table II and shown in figures 2 through 6. The data glitches that occurred at approximately 12 148 seconds on these figures were caused by a loss of telemetry for a brief period of time, and do not represent an anomalous behavior of the propulsion system. The propellant densities used were calculated from the equations presented in the section on propellant loading analysis for an assumed temperature of 70° F, and for the flight inlet pressures. The estimated spacecraft damp weight was obtained from the Apollo Spacecraft Program Office. The initial estimates of the propellants on board at the beginning of the time segment analyzed were extrapolated from the loaded propellant weights discussed in the section on propellant loading analysis.

The results of the propulsion analysis program simulation of the 300-second burn segment are contained in table III and shown in figures 7 through 10. The values presented in table III represent results midway between FS-1 and storage tank depletion for the "before crossover" values, and midway between storage tank depletion and FS-2 for the "after crossover" values. The results are considered representative of the actual flight values throughout these portions of the burn. Before storage tank depletion, thrust was between 20 829 and 20 851 pounds; specific impulse was between 310.1 and 310.2 seconds; and mixture ratio varied from 1.983 to 1.990. Following storage tank depletion, thrust was between 21 342 and 21 373 pounds; specific impulse was between 310.0 and 310.3 seconds; and mixture ratio varied from 1.996 to 2.008.

The time history of the measured chamber pressure (SP0661P) during the SPS burn is shown in figure 2. Prior to, and following the SPS burn, the measurement read approximately 2 psi below zero, which indicated a bias probably existed in the steady-state reading during the burn. A comparison between the measured chamber pressure and the program-computed chamber pressure is shown in figure 11. The residual error (measured — computed) shown in figure 11 was adjusted to account for the indicated bias. However, as shown by the trend in the residual error in figure 11, the measured chamber pressure exhibited an upward drift during the burn which was not substantiated by the analysis. A

similar drift has been observed on all previous SPS flights, and is presently attributed to thermal effects from the combustion chamber which is not uncommon.

The program simulation also indicated that the measured fuel inlet pressure (SP0010P) read 5.6 psi low throughout the burn. The fuel check valve outlet pressure (SP0006P) was also determined to be reading approximately 20.0 psi low for the major portion of the burn.

Shown in figures 11 through 18 are analysis program output plots which represent the residual errors or differences between the filtered flight data and program-calculated values. Also presented on these figures are the filtered flight data. These figures represent chamber pressure, thrust acceleration, oxidizer sump tank gaging system, fuel sump tank gaging system, oxidizer storage tank gaging system, and fuel storage tank gaging system, respectively. A strong indication of the accuracy of the analysis program simulation can be obtained by comparing the thrust acceleration calculated in the simulation to that derived from the Apollo guidance computer (AGC) ΔV date transmitted via measurement CG0001V. Figure 12 shows the thrust acceleration during the portion of the burn analyzed as derived from the AGC data, and the residual error between the AGC and program-calculated values. The residual error time history has essentially a zero mean, and little, if any, discernible trend. This indicates the simulation is relatively valid, although other factors must also be considered in critiquing the simulation.

Because no propellant temperature telemetry measurements were available, a confident determination of propellant densities was not possible; which, in turn, reduces confidence in the simulation results. The lack of point sensor data (auxiliary propellant utilization and gaging system) also compromises the simulation somewhat by magnifying one of the most difficult analysis problems, that of determining propellant flow rates.

Critique of Preflight Performance Prediction

Prior to the Apollo 6 mission the performance of the SPS was predicted in the MSC Internal Note MSC-EP-R-68-6, Apollo Service Propulsion System Preflight Analysis, dated March 22, 1968. This performance prediction was for the integrated propellant feed/engine system which was characteristic of the SPS hardware on this flight. Thus, it was a preflight estimate of the propulsion system performance under the space flight conditions, with no restrictions placed upon conditions at the inlet to the engine.

The predicted performance parameters are compared to the analysis program-calculated inflight performance parameters in figures 7 through 9.

The differences between the flight values and the predicted values are due partially to the fact that the predictions were based upon a combustion chamber performance characterization generated from engine class data, which have been acceptable in the past because of the small variations from engine to engine. The SC 020 engine, however, was reported in the acceptance test log to be a low-performing engine. Since this was the first engine to deviate significantly from the class performance, there was a question as to whether the class characterization from the altitude tests or the performance value obtained from the sea-level acceptance test of this particular engine should be used. In this particular case it was decided to use the class characterization. However, the results of the flight and subsequent analysis indicate that this general characterization should be adjusted to reflect the acceptance test values of each engine.

As shown in figure 7, the analysis program-calculated flight thrust was essentially constant following storage tank depletion. A similar flight thrust trend was observed on the Apollo 4 mission. The predicted thrust, however, decreased, at close to a constant rate or approximately 125 pounds over the same time period. Analysis has indicated that the flight analysis program thrust trend is more representative of the actual thrust, and that the predicted decrease was caused by inaccurate modeling of the helium regulator. Future predictions will account more accurately for the increase in regulator outlet pressure associated with the decrease in regulator inlet pressure which occurs as the helium supply bottle is depleted.

Engine Performance at Standard Inlet Conditions

The engine acceptance tests are conducted with controlled engine/system interface conditions. This enables the engines to be evaluated on their own merit and provides a common basis for comparison of engines. It was determined from the analysis of the Apollo 6 flight that the SPS engine performance corrected to standard inlet conditions yielded a thrust of 21 357 pounds, a specific impulse of 309.8 seconds, and a propellant mixture ratio of 2.014. The values of thrust, specific impulse, and mixture ratio are 0.31, 0.03, and 0.20 percent higher, respectively, than the standard inlet condition values reported in the acceptance test log for test no. 3.5-07-DPA-043 on engine S/N 0000032. The differences are within the accuracy with which these values can be determined. The standard inlet conditions performance values reported herein were calculated for the following conditions.

Condition	Performance value
Oxidizer interface pressure	164 psia
Fuel interface pressure	170 psia
Oxidizer interface temperature	70° F
Fuel interface temperature	70° F
Oxidizer density	90.15 lbm/ft^3
Fuel density	56.31 lbm/ft^3
Thrust acceleration	1.0g
Throat area (initial value)	121.56 in^2

GAGING SYSTEM ANALYSIS

The propellant utilization and gaging system (PUGS) was operated in the primary mode. The storage and sump tank mass data were individually transmitted on separate measurements.

All gaging system signals were locked on preset values for a brief period following SPS ignition to prevent excessive oscillations from propellant slosh during the no-ullage start. The expected lockout period was 4.5 seconds. Because of the lockout period both the fuel and oxidizer storage tank gages indicated an excessively high flow rate immediately following the end of the lockout period. A data spike observed in the fuel storage tank gage reading near ignition is attributed to a shorter than expected lockout period on that gage and propellant oscillations associated with the no-ullage start. North American Rockwell (NR) reported that a circuit analysis showed the lockout period on the fuel gage to be 2 seconds and 4 seconds for the oxidizer gage. After stabilization, the depletion rates indicated by the storage tank gages were within 0.9 and 0.21 percent of the computed values for oxidizer and fuel flow rate, respectively. The oxidizer storage tank gage reading, when extrapolated to ignition, was not consistent with the prelaunch reading. The extrapolation showed an equivalent reading of approximately 6280 pounds, compared to a prelaunch reading, as noted in the section on propellant loading of 6520 pounds under 175-psia tank

pressure. This difference can be attributed to either a shift of the oxidizer from the storage tank to the sump tank due to helium going into solution during boost, or a shift in the gaging system's calibration. Neither can be substantiated since the sump tank liquid level was above the maximum sensing point of the gaging system. The oxidizer storage tank gage also showed a +100 bias at depletion. This magnitude of bias was also noted on the Apollo 4 mission, and is now believed to be associated with improper description of the zero-point calibration of the storage tank probe in the loading procedures.

Prior to storage tank depletion (crossover) both the oxidizer and fuel sump tank gages indicated a small continuous rise in level. A known bias exists in the sump tank gage readings because of difference in liquid levels in the sump tanks and the gaging system stillwells. The stillwell is a manometer which balances the pressure at the bottom of the stillwell with a fluid head. Under nonflow conditions, this fluid head is equivalent to the level of the propellant in the tank. However, when the propellant is flowing, the fluid head in the stillwell is reduced by the dynamic head of the propellant flowing by the bottom of the stillwell through the zero-g retention reservoir. Because of the lockout period, and the fact that the ground propellant levels at 175-psia tank pressure are over the sensing elements of the probes, it is difficult to determine the exact bias effect from flight data. Following the lockout period there is an apparent drop in the sump tank levels. This is caused by the decrease in levels inside the stillwells which results from the flow effects discussed above. The indicated continuous rise in sump tank levels during the time prior to crossover is caused by changes in the dynamic flow bias with acceleration, as was expected.

The levels start to decrease in the sump tanks within 2.0 seconds of the crossover time as determined from the rise in engine inlet pressure and storage tank depletion. As observed on the Apollo 4 mission, there is an indicated high flow rate for the first 10 to 15 seconds following crossover in both oxidizer and fuel. This condition is caused by the fact that initially the levels in the sump tanks are above the cylindrical section, in the hemispherical part of the tank. Because of the dynamic flow bias, the probe senses a lower level which, based on the shaping of the probe, is associated with a larger tank diameter. Since the probe is really sensing a height change, the apparent flow rate is high until the levels reach the cylindrical section of the tanks. After stabilization, the sump tank gages showed a normal depletion rate.

Analysis of the flight data and investigations into prelaunch procedures pointed out two problems with the PUGS data. As previously mentioned, the failure of the oxidizer storage tank gage reading to go to zero at tank depletion is believed to be associated with the zero-point calibration of the probe prior to launch. The storage tank

oxidizer probe was zero point calibrated prior to launch with no oxidizer in the tank. North American Rockwell confirms that the Block I oxidizer capacitance probe should be zero point calibrated with oxidizer covering the dielectric compensator at the bottom of the probe. However, the exact nature (scale factor or bias) of the resulting error and its magnitude are not presently known. It is observed, however, that the largest difference between the analysis program-computed flow rates and the PUGS-indicated flow rates was in the oxidizer storage tank.

Another PUGS data problem was revealed when it was determined that the PUGS calibration curves (equations) used by MSC data reduction to reduce the flight telemetry data may not have been the same calibration curves as used in the automatic checkout equipment (ACE) at KSC when the PUGS was prelaunch calibrated. The MSC data reduction calibration curves were generated from the PUGS acceptance test data. Informal information from KSC indicates the ACE calibration curves were nominal calibrations; that is, 0 through 7500 pounds for fuel and 0 through 15 000 pounds for oxidizer. Although there were significant differences between the two sets of calibration curves, use of the reported ACE curves in the analysis program did not significantly affect the computed performance.

Because of the unresolved PUGS data problems it is not possible to evaluate realistically the PUGS accuracy to the degree required by the test objectives success criteria as discussed in the section on detailed test objectives. However, the PUGS did appear to function properly, and no gross errors in quantity were observed. Block II PUGS procedures and operation are presently being reviewed to prevent these problems from affecting future missions.

PRESSURIZATION SYSTEM ANALYSIS

The SPS helium pressurization system operated nominally throughout the mission. There was no indication of leakage. Helium bottle pressure and temperature indicated nominal helium expulsion during the entire burn. The resulting ullage pressures and changes in propellant volumes obtained from the analysis program for the 300 seconds of burn time analyzed indicated that 32.9 pounds of helium were added to the ullages. For the same time period the helium bottle pressure and temperature measurements (figs. 19 and 20, respectively) indicated a decrease of 31.8 pounds of helium from the supply bottles. The difference in the values is considered within the instrumentation accuracy of the helium bottle measurements.

Pressure oscillations were experienced in the helium pressurization system for the first 10 seconds of the burn. The oscillations were seen

in the helium line upstream of the regulator (SP0001) and in the helium lines below the check valves (SP0003 and SP0006). Propellant pressure measurements at the engine inlet indicate that the oscillations were completely damped in the ullages. The oscillations were caused by the initial propellant tank pressures being higher than the regulation pressures. It is a characteristic of the regulator to oscillate when the demand is below the rated value.

ENGINE TRANSIENT ANALYSIS

An analysis of the start and shutdown transients was performed to determine the transient impulse and time-variant performance characteristics during the Apollo 6 mission and to ascertain the effectiveness of the no-ullage start.

The results of this analysis are summarized in table IV. Engine acceptance test data, specification requirements, and previous spacecraft flight data were employed to provide better insight into the significance of the Apollo 6 flight test results and the applicability thereof to subsequent flight development missions and to the lunar landing mission. Start and shutdown transient plots of chamber pressure are shown in figures 21 and 22.

All applicable transient specification criteria appeared to be satisfied, except for the chamber pressure overshoot and engine start impulse. The chamber pressure overshoot has also occurred during other flights and will be discussed in depth later in this section, as will the start impulse.

The techniques utilized in evaluating the SPS transient performance and behavior characteristics during the Apollo 6 mission are detailed in the ensuing text.

The velocity gain for the single SPS engine burn cut-off (c/o) was calculated to be 15.30 ft/sec referenced to a cut-off time of 12 207.93 seconds g.e.t. from guidance system data. The estimated average vehicle weight at this time was 25 036 lbm. The c/o impulse is defined as the thrust-time integral as follows

$$I = \int_{t_{c/o}}^{t_{F=0}} F dt \tag{1}$$

Inserting $F = ma/g_c$ and assuming the mass is approximately constant during cut-off, the following is obtained

$$I = \frac{m}{g_c}\int_{t_{c/o}}^{t_{F=0}} a\,dt = \frac{-m}{g_c}\left(V_{t_{c/o}} - V_{t_{F=0}}\right) \qquad (2)$$

where I = cut-off impulse, lbf-sec

F = thrust, lbf

t = time, sec

m = total vehicle mass, lbm

a = thrust acceleration, ft/sec^2

g_c = conversion factor, lbm-ft/lbf-sec^2

V = thrust velocity, ft/sec

From equation (2) the cut-off impulse for the burn can be calculated as follows

$$I = \frac{25\ 035}{32.174} \times 15.30 = 11\ 905.4 \text{ lbf-sec}$$

The time from the cut-off signal to thrust equals zero was 1.42 seconds.

To obtain the impulse from FS-2 (c/o signal) to 10-percent thrust, the integral of the chamber pressure P_c between those limits was used. The relation used is as follows

$$I = \int_{t_{c/o}}^{t_{F=10\%}} F\,dt = \int_{t_{c/o}}^{t_{F=10\%}} C_f P_c A_t\,dt = C_f A_t \int_{t_{c/o}}^{t_{F=10\%}} P_c\,dt \qquad (3)$$

where C_f and A_t are assumed constant during the transients

C_f = thrust coefficient, unitless

A_t = throat area, in^2

P_c = chamber pressure, lbf/in^2

In calculating the transient impulse, the value of C_f which is often used when transient thrust data are not available, is the steady-state C_f value. The actual value of C_f is influenced by the mixture ratio and the chamber pressure, both of which are rapidly changing during the start and cut-off periods. Therefore, to improve the estimate of the transient impulse, the value of C_f used for the other transients was determined by applying equation (3) to the cut-off impulse determined by the velocity gain.

$$C_f = \frac{I}{A_t \int_{t_{c/o}}^{t_{F=0}} P_c\,dt} = \frac{11\ 905.4}{123.99 \times 55.2468} = 1.738$$

Applying equation (3) the following result was obtained for the cut-off impulse from FS-2 to 10-percent steady-state thrust.

$$I = 1.738 \times 123.99 \times 53.9526 = 11\ 626.5 \text{ lbf-sec}$$

where

$$\Delta t \Big|_{FS-2}^{F=10\%} = 0.90 \text{ sec}$$

Since velocity gain is measured at a very low sample rate (1 sample per 2 seconds), the velocity gain could not be readily determined during the start impulse and equation (3) was again used.

Applying equation (3) the following results were obtained. Transient start impulse from FS-1 to 90-percent steady-state thrust

$$I = 1.738 \times 121.56 \times 1.0650 = 225.0 \text{ lbf-sec}$$

where

$$\Delta t \Bigg|_{FS-1}^{F=90\%} = 0.46 \text{ sec}$$

Transient start impulse from FS-1 to steady-state thrust

$$I = 1.738 \times 121.56 \times 88.9115 = 18\ 784.4 \text{ lbf-sec}$$

where

$$\Delta t \Bigg|_{FS-1}^{F=100\%} = 1.25 \text{ sec}$$

The impulse contained in the chamber pressure overshoot was determined to be 1908 lbf-sec or 10.16 percent of the total start impulse.

As shown in table IV, the SPS data summary, the start transient impulse did not meet specification values as delineated in the CSM Master End Item Specification, SID 64-1237. The engine did, however, perform as expected based on the acceptance test data, and thus satisfied the specifications applying to acceptance tests. It should be resolved which specification should be used for flight evaluation. The transient values compared favorably with the data acquired from previous flight tests.

As can be seen in figure 19, the chamber pressure transient overshoots the steady-state value by approximately 50 psi. This behavior

has also been noted on previous flights. Table V is a summary of the past flight data.

The first two columns in this table identify the mission, followed by columns of the four shutoff valve opening times and the chamber pressure overshoot values. The remainder of the data consists of the initial valve inlet pressure drop that occurs when the valves are opened and the available data regarding the valve actuation source pressure. It can be seen from the 202 and 501 data that successive starts reduce the overshoot.

Since the chamber pressure is sampled at 100 samples per second and because the range of the telemetry is 0 through 150 psia, the true value of the measured overshoot may not be recorded. An investigation of ground and flight test data has been conducted to determine whether the indicated overshoot is partially due to instrumentation inaccuracies or whether it is totally characteristic of the start. The conclusions that can be made regarding the overshoot as determined by the ground test study are the flight transducer is sensitive to thermal effects and that the flight transducer amplifies the actual pressure overshoot from 10 to 40 percent. Attempts at correlating the flight transducer indicated overshoot to the actual overshoot have been unsuccessful.

DETAILED TEST OBJECTIVES

As an integral part of the system evaluation processes necessary to certify the SPS for subsequent manned flights and/or lunar landing missions, detailed test objectives were outlined for the Apollo 6 mission. The objectives provided a systematic opportunity for acquisition of data to certify the SPS for manned operations and subsequent lunar missions. Objectives peculiar to the propulsion system were P3.2 (SPS No-Ullage Start) and P3.3 (SPS Long-Duration Burn).[1] Both of the objectives were established as secondary objectives for the Apollo 6 mission because they had previously been primary objectives for the Apollo 4 mission.

An examination of the Apollo 6 postflight analysis results was made in view of the success criteria specified for the two objectives. An explanation of each test objective and the degree to which its success criteria were satisfied is presented in the following discussion.

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[1]NASA Report, "Apollo 4 and 6 Mission Requirements (501/017/LTA-10R and 502/020/LTA-2R), Unmanned Supercircular Reentry," dated September 27, 1967.

Objective P3.2 (SPS No-Ullage Start)

Objective P3.2 was intended to demonstrate that an SPS start can be satisfactorily performed in a near zero-g environment without an RCS ullage maneuver when the sump tanks are full. This mode of operation is dictated by the necessity of conserving RCS propellants. Five criteria were specified in the mission requirements for determining the success or failure of the objective. The requirements and the related postflight analysis results are presented in table VI. Postflight analysis indicated that the success criteria for the objective, as written, were only partially met. Success criteria no. 1 called for the start impulse to be between 400 through 1200 lb-sec. The flight value (see section on engine transient analysis) of 225 lb-sec was outside this band, but was equal to the acceptance test value and close to the values on previous flights.

The propellant interface pressures specified in success criteria no. 3 were not measured. The values presented in table VI were calculated from the measured engine valve inlet pressures corrected for the indicated biases. The specified values for the interface pressures of 163 ± 4 psia for fuel and 160 ± 4 psia for oxidizer are not consistent with the established Block I nominal values of 170 psia and 164 psia, respectively, and do not allow for the expected rise in interface pressures at crossover. The calculated values of both interface pressures during the Apollo 6 mission SPS burn were within approximately 3 psi of predicted and are considered satisfactory.

The failure of the system to meet success criteria no. 5, the maximum chamber pressure overshoot, is not considered to be directly related to the no-ullage start. Chamber pressure overshoots have been experienced on all previous SPS flights and are presently attributed to valve opening times and instrumentation response characteristics.

The flight test data showed no symptoms of helium ingestion, demonstrating the effectiveness of the propellant retention screens and the zero-g retention reservoirs in maintaining propellants over the sump tank outlets (feedline inlets).

Objective P3.3 (SPS Long-Duration Burn)

Objective P3.3 had as its purpose the determination of the effect of burn duration on SPS performance. Four success criteria were specified in the mission requirements for the evaluation of the success or failure of the objective. The four success criteria and the respective flight test values are presented in table VII.

The deviations between flight results and success criteria no. 1 were previously covered in the discussion of the no-ullage start objective. The PUGS accuracy required in success criteria no. 3 was not demonstrated by the flight data (see section on gaging system analysis); however, calibration and data reduction inconsistencies made the PUGS accuracy determination inconclusive.

CONCLUSIONS

The performance of the Spacecraft 020 service propulsion system during the Apollo 6 mission was based on a 300-second segment of the 442-second burn. Since the remaining data showed no anomalies, the reported values are considered representative of the entire burn. The results of the analysis of the flight data are considered to be as accurate as the available instrumentation and associated instrumentation accuracies will permit. Therefore, no further analysis is anticipated.

The non-hardware-associated problems which were presented in this report are as follows:

1. Propellant density determination
2. Propellant loading procedures
3. Instrumentation errors
4. Preflight modeling errors

These problems are being studied at present.

As mentioned in the report the fuel density reported by the Bendix laboratory is thought to be in error. It is felt that a more accurate value could be achieved by requiring more than one sample of the propellants be taken. This requirement should be incorporated into the Block II loading procedures, which currently are being formulated.

Several problems associated with the propellant loading procedures were encountered. The zero point of the oxidizer storage tank probe was adjusted incorrectly resulting in a bias and possibly a scale factor error in the output of this probe. The magnitude of the error is unknown, but it could be significant enough to affect the propellant

unbalance logic associated with the use of the propellant utilization valve. This could result in an improper propellant tank mixture ratio control. The Block II loading procedures should be investigated to prevent similar problems on future missions. Also pointed out in the section on propellant loading analysis were the discrepancies in the KSC reported propellant loads. This problem is being resolved and corrections are being incorporated into the Block II procedures. Discussed in the gaging systems analysis sections were the incorrect calibration curves used in the MSC data reduction of the gaging system flight data. A requirement should be implemented for supplying the correct calibration curves to MSC for the reduction of the flight data.

Errors apparently present in the instrumentation on Spacecraft 020 can partially be eliminated by using calibrations for the particular transducer and signal conditioner on the flight article. At present, class-type calibrations are being used and no attempt is being made to remove known biases. This problem is currently being discussed with the instrumentation personnel in the Apollo Spacecraft Program Office.

Two problems are associated with the preflight predictions as discussed in the steady-state performance analysis section. First, the preflight prediction of the Spacecraft 020 performance was based upon combustion chamber performance from engine class data and did not take into account the reported low performance from the acceptance test of this engine. As indicated by the analysis of this flight, the acceptance test data should influence the preflight predictions. In the future the preflight performance predictions will take into account the engine acceptance test data. Secondly, the trends that the performance parameters exibited in the preflight predictions did not agree with the trends of these parameters as determined by the postflight analysis of this flight. A similar discrepancy was observed on the Apollo 4 mission. This problem is presently believed to be associated with the helium regulator model used in the preflight prediction program. A more detailed investigation will be made.

There was one potential problem that can possibly be related to the hardware on Spacecraft 020. This problem is the chamber pressure overshoot which was discussed in the transient analysis section. Ground tests of the SPS starts are being conducted to investigate this problem.

In summary, the overall performance of the Apollo 6 service propulsion system was considered to be nominal and the two detailed test objectives, no-ullage start and long-duration burn, were accomplished to a satisfactory degree.

TABLE I.- APOLLO 6 MISSION DATA FILTER SPANS

Measurement	Time period, g.e.t., sec					
	11805-11884	11885-11904	11905-11910	11911-11919	11920-11934	11935-12105
SP0009P	10.0	10.0	5.0	10.0	10.0	10.0
SP0010P	10.0	10.0	5.0	10.0	10.0	10.0
SP0661P	10.0	10.0	(a)	10.0	[illegible].0	10.0
SP0655Q	10.0	10.0	(a)	(b)	(b)	(b)
SP0656Q	10.0	10.0	(a)	(c)	10.0	10.0
SP0657Q	10.0	10.0	(a)	(b)	(b)	(b)
SP0658Q	10.0	10.0	(a)	(c)	10.0	10.0
Acceleration	18.0	2.5	(a)	8.0	8.0	18.0

[a]Deleted during this period due to the occurrence of crossover.

[b]Deleted during this period since the measurement should be reading zero.

[c]Deleted during this period for problems associated with modeling the noncylindrical section of the propellant tank.

TABLE II.- FLIGHT DATA USED IN THE ANALYSIS PROGRAM

Measurement number	Description	Nominal data range	Sample rate, samples/sec
SP0009P	Main valve, engine oxidizer inlet pressure	0 through 300 psia	10
SP0010P	Main valve, engine fuel inlet pressure	0 through 300 psia	10
SP0655Q	Oxidizer tank no. 1 primary quantity	0 through 16 000 lb	1
SP0656Q	Oxidizer tank no. 2 primary quantity	0 through 16 000 lb	1
SP0657Q	Fuel tank no. 1 primary quantity	0 through 8000 lb	1
SP0658Q	Fuel tank no. 2 primary quantity	0 through 8000 lb	1
SP0661P	Engine chamber pressure	0 through 150 psia	100
CG0001V	Computer digital data	40 bits	1/2

TABLE III.- APOLLO 6 MISSION SPS PERFORMANCE SUMMARY

Conditions	Value				
	Nominal	Before crossover		After crossover	
		Measured	Actual[a]	Measured	Actual[a]
Measurement description					
SP0003 — Oxidizer storage tank pressure, psia	≈179	174	181.1	176	182.[illegible]
SP0006 — Fuel tank storage pressure, psia	≈179	168	182.3	162	18[illegible].5
SP0009 — Oxidizer inlet pressure, psia	≈154	154	154.8	161	161
SP0010 — Fuel inlet pressure, psia	≈154	147	152.3	152	157.5
SP0661 — Engine chamber pressure, psia	≈100	97	99.5	102	102.6
Calculated performance parameters					
Oxidizer flow rate, lbm/sec . . .	≈45.8	--	44.67	--	45.89
Fuel flow rate, lbm/sec	≈22.9	--	22.51	--	22.95
Propellant mixture ratio	2.00 (±1 percent)	--	1.984	--	2.000
Vacuum specific impulse, sec . . .	313 minimum	--	310.2	--	310.3
Vacuum thrust, lbf	21 500 (±1 percent)	--	20 840	--	21 360

[a]Program analysis results for the flight data not corrected to standard inlet conditions.

TABLE IV.- SERVICE PROPULSION SYSTEM FLIGHT SUMMARY OF TRANSIENT DATA

	AS-502 burns		AS-501 burns		AS-202 burns				Specification values
	1st	SPS Engine 032 acceptance test	1st	2nd	1st	2nd	3rd	4th	
Start transient total vacuum impulse from FS-1 to 90 percent of steady-state thrust, lbf-sec	225.0	225	135.2	264.4	--	--	--	--	[a]400-1200 [b]100-400
Time from FS-1 to 90 percent of steady-state thrust, sec	0.46	0.375	0.41	0.35	0.38	~0.37	~0.37	~0.37	[a]0.40-0.60 [b]0.350-0.550
Engine run-to-run start repeatability, lbf-sec	--	--	200 ± 65	200 ± 65	--	--	--	--	[b]±100
Start transient total vacuum impulse from FS-1 to steady-state thrust, lbf-sec	18 784.4	--	15 852.0	14 173.2	--	--	--	--	--
Time from FS-1 to steady-state thrust, sec	1.25	--	1.07	0.96	~1.09	~1.18	~1.25	~1.14	--
Percent chamber pressure overshoot during start, percent	58.	--	56.0	40.0	41	19	22	--	20
Percent of start impulse (FS-1 to steady-state) contained in chamber pressure overshoot	10.16	--	11.74	--	--	--	--	--	--
Shutdown transient total vacuum impulse from FS-2 to 10 percent of steady-state thrust, lbf-sec	11 626.5	8082	10 083.6	11 910.8	--	--	--	--	8000-13 000
Time from FS-2 to 10 percent of steady-state thrust, sec	0.90	0.687	0.80	0.89	~1.049	~1.11	~0.78	~0.78	[a]0.40-0.060 [b]0.650-0.900
Engine run-to-run shutdown repeatability, lbf-sec	--	--	10 997.3 ± 914	10 997.3 ± 914	--	--	--	--	[b]±100
Shutdown transient total vacuum impulse from FS-2 to 0 percent thrust, lbf-sec	11 905.4	--	11 122.3	12 275.7	10 700	--	--	10 000	[b]8830-14 200
Time from FS-2 to 0 percent thrust, sec	1.416	--	1.82	1.50	--	--	~1.23	~1.13	--

[a]CSM Master End Item Specification, SID 64-1237.

[b]SPS Engine 032 Acceptance Test Report.

TABLE V.- SERVICE PROPULSION SYSTEM VALVE RESPONSE AND PRESSURE OVERSHOOT CHARACTERISTICS

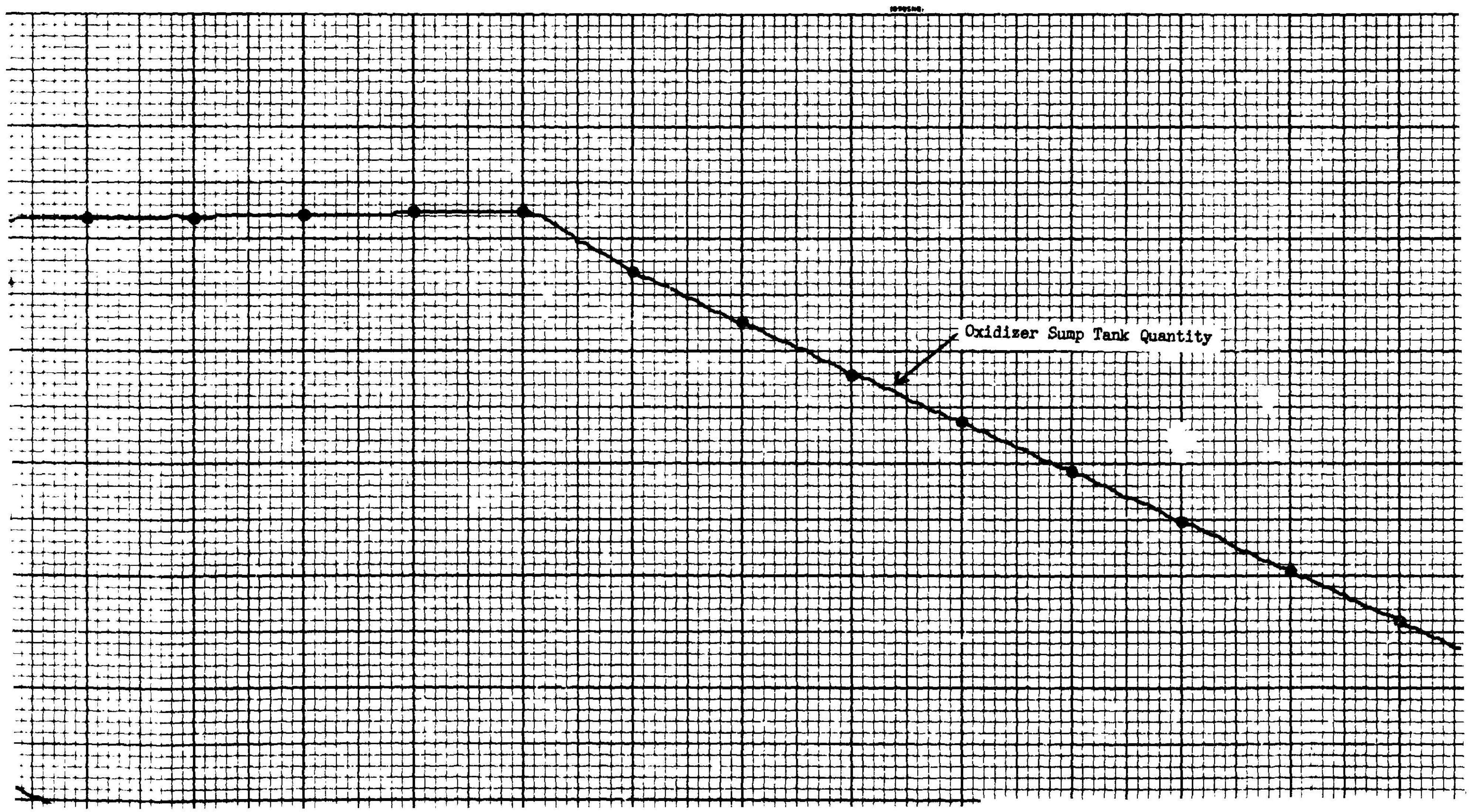
Mission	Burn	Bipropellant valve opening time, sec				Maximum chamber pressure overshoot		Maximum drop in valve inlet pressure		Engine valve actuated system tank pressure			
										SP0600 primary, psi		SP0601 secondary, psi	
		1st	2nd	3rd	4th	psi	Percent	Oxidizer	Fuel	Start	Cut-off	Start	Cut-off
AS-201	1st	0.62	0.41	0.31	0.61	132	35	50	82	812	770	(a)	(a)
	2nd	.64	.43	.34	.44			53	57	770	727	(a)	(a)
AS-202	1st	0.60	0.20	0.30	0.60	136.5	41	54	86	1608		2435	
	2nd	.70	.28	.30	.80	122	19	41	51				
	3rd	.80	.30	.40	.82	124.5	22	40	73				
	4th	.80	.80	.30	[illegible]	111	9	38	72				
AS-501	1st	0.60	0.30	0.30	0.50	148.6	56	58	82	2475	2475	2525	2525
	2nd	.65	.60	.50	.50	134.2	40	40	51				
AS-502	1st	0.70	0.40	0.40	[illegible]	156	58	48	88.5	2510	2450	2530	2470

[a]Bad Data — recorded values were approximately 15 psia.

TABLE VI.- SERVICE PROPULSION SYSTEM NO-ULLAGE START OBJECTIVE

Success criteria	Test value
1. Start transient total impulse from onset of electrical command to 90-percent steady-rated thrust must be within Master End Item Specification SID 64-1237, paragraph 3.4.1.3.4.1.4.7.2. The start transient total impulse from onset of electrical command to 90-percent rated thrust shall be from 400-lb-sec (minimum) to 1200 lb-sec (maximum).	225 lb-sec
2. The engine must develop 90-percent steady-state thrust within 0.4 to 0.6 second after onset of the electrical command signal to the pilot valve.	0.46 second
3. During starting, the fuel and oxidizer pressures are within 6 psi of each other. During steady-state engine operation, the fuel is furnished to the propellant interface (PIF) at 163 ± 4 psia and the oxidizer is furnished to the propellant interface (PIO) at 160 ± 4 psia.[a]	ΔP = 5 psia PIO = 1[illegible]7.5 through 157.9 psia } Before PIF = [illegible]7 through 165.4 psia } crossover PIO = [illegible].5 through 164.2 psia } After PIF = 170.0 through 171.0 psia } crossover
4. The steady-state thrust and mixture ratio (MR), extrapolated to reflect nominal engine value inlet propellant supply conditions, are to be within ±1 percent of 21 500 pounds and 2.00, respectively.	Thrust = 21 35[illegible] MR = 2.014
5. The transient starting chamber pressure P_c is not greater than 120 percent of nominal chamber pressure.	Maximum $P_c \approx$ 150 percent of nominal

[a]Not consistent with Block I nominal values of 170 psia for fuel and 164 psia for oxidizer.


Oxidizer Sump Tank Quantity

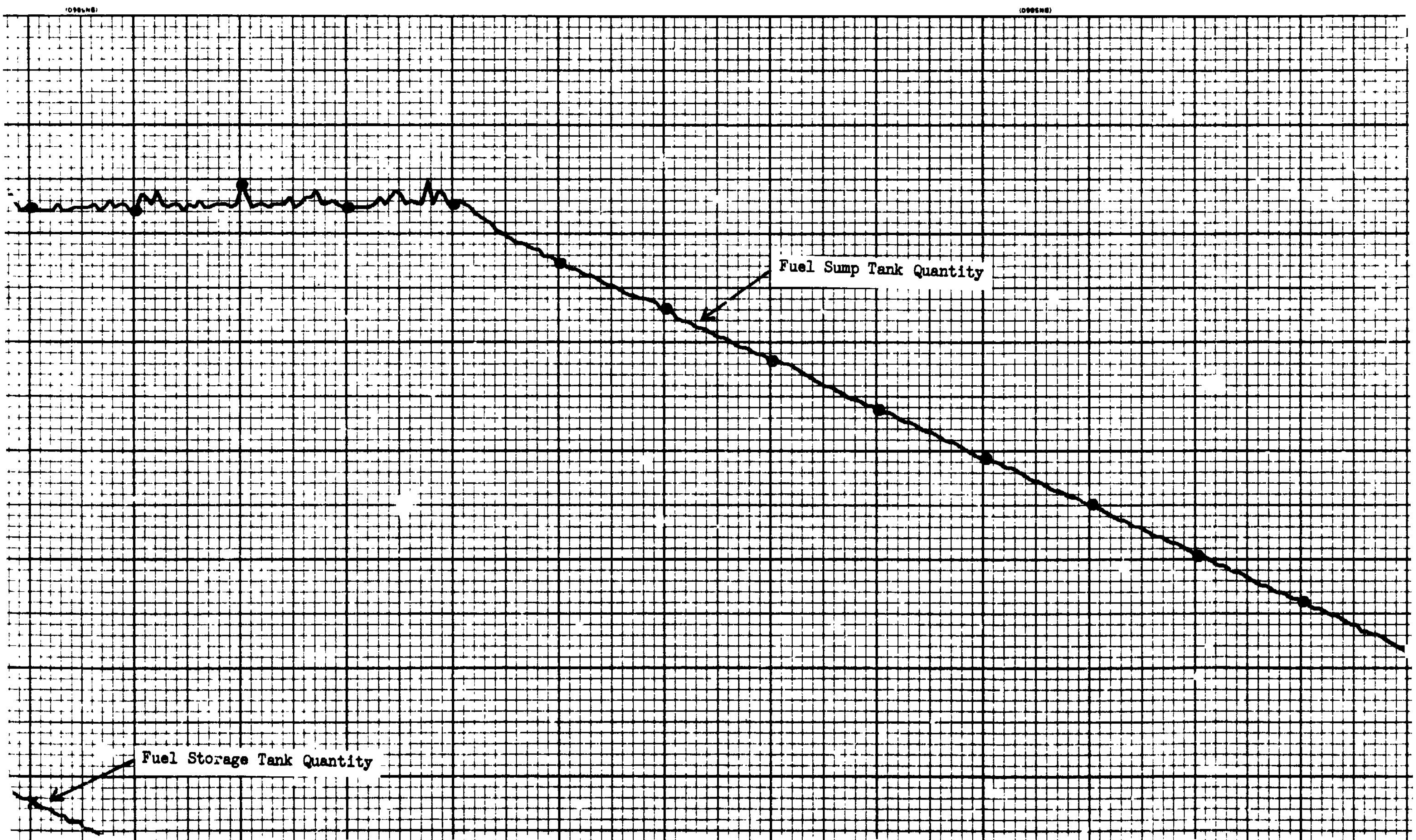

Fuel Sump Tank Quantity
Fuel Storage Tank Quantity

TABLE VII.- SERVICE PROPULSION SYSTEM LONG-DURATION BURN OBJECTIVE

Success criteria	Test value
1. During steady-state engine operation, the fuel is furnished to the propellant interface (PIF) at 163 ± 4 psia and the oxidizer is furnished to the propellant interface (PIO) at 160 ± 4 psia.[a]	PIO = 157.5 through 157.9 psia, PIF = 164.7 through 165.4 psia: Before crossover PIO = 163.5 through 164.2 psia, PIF = 170.0 through 171.0 psia: After crossover
2. The steady-state thrust and mixture ratio (MR), extrapolated to reflect nominal engine valve inlet propellant supply conditions, are to be within ±1 percent of 21 500 pounds and 2.00, respectively.	Thrust = 21 357 pounds MR = 2.014
3. Propellant utilization gaging system (PUGS) accuracy (after correction for PUGS bias) must be within 0.35 percent of full tankage capacity plus 0.35 percent of propellant remaining (applies separately to oxidizer and fuel).	(See discussion)
4. Shutdown impulse is to be within 8000 through 13 000 lb-sec.	11 626 lb-sec

[a]Not consistent with Block I nominal values of 170 psia for fuel and 164 psia for oxidizer.

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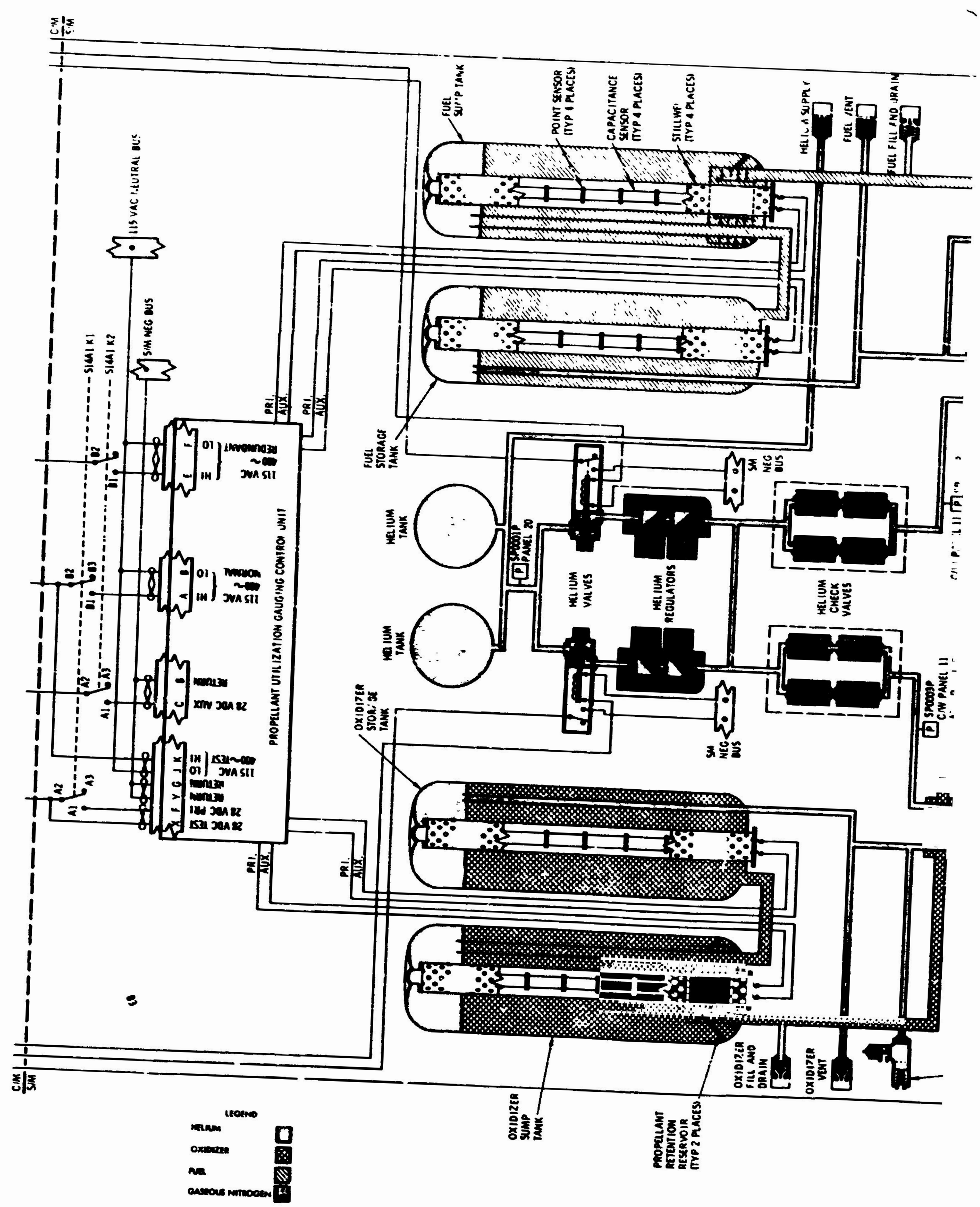

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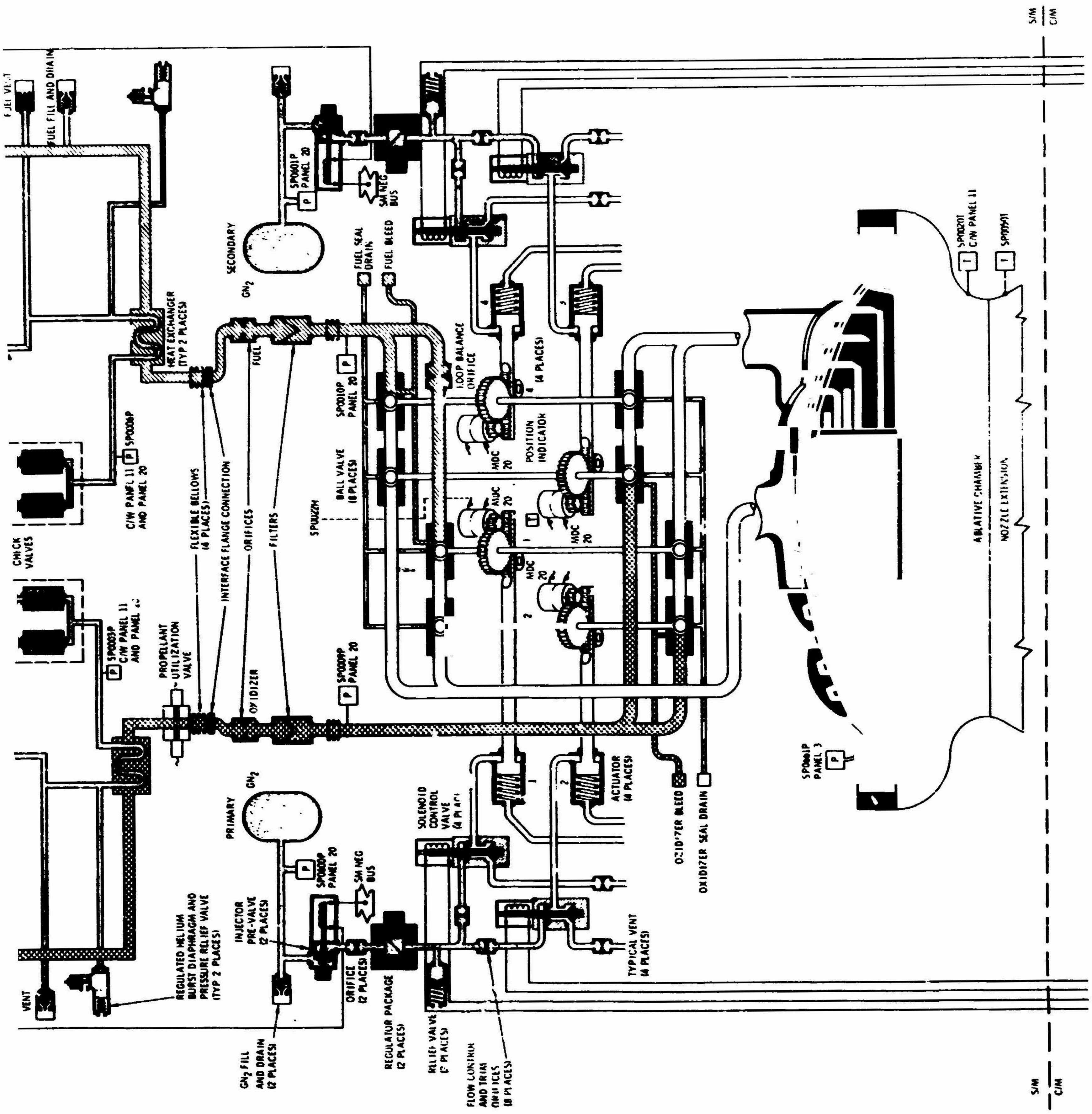

Figure 1.- Service propulsion system functional flow diagram.

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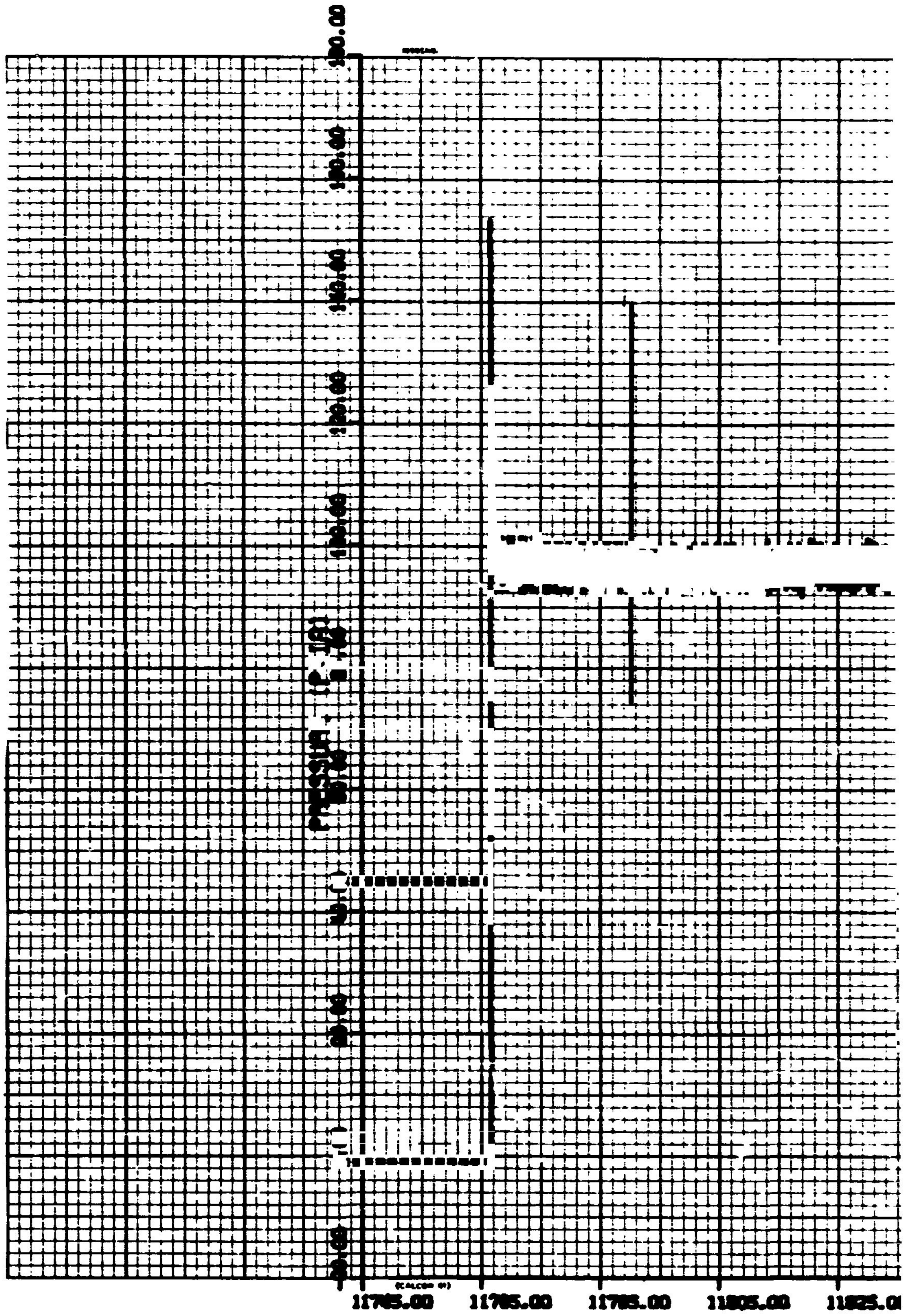

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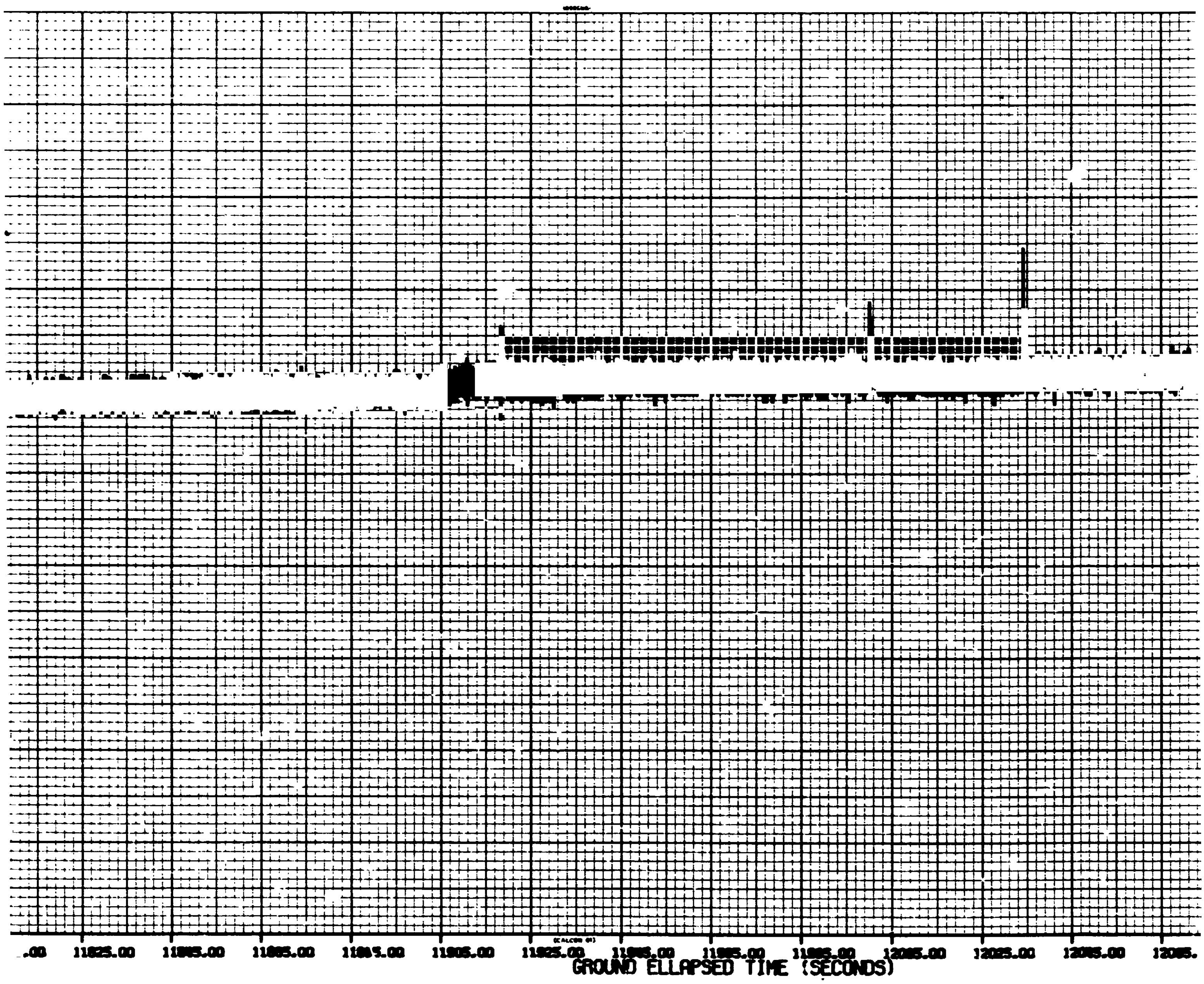

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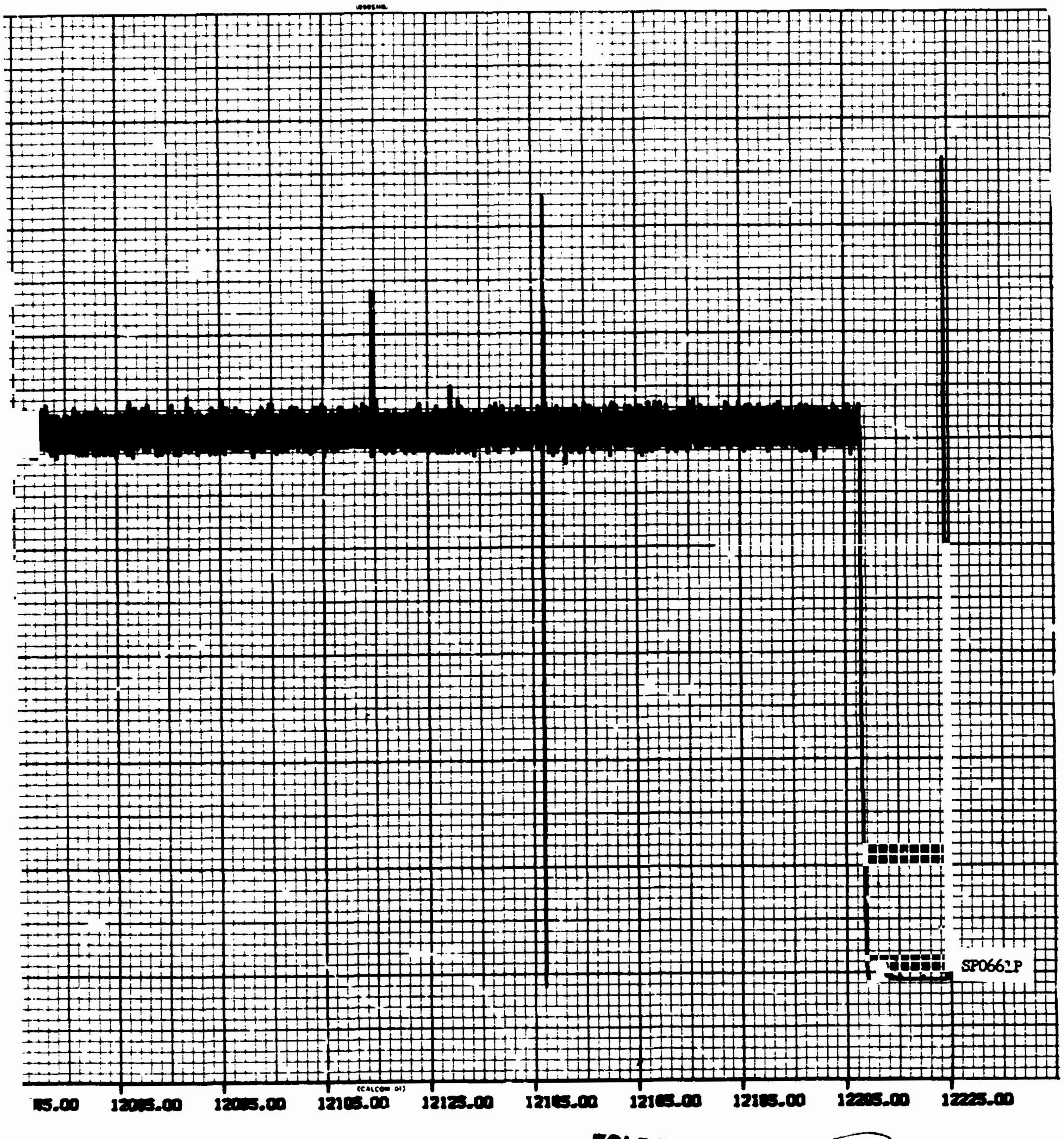

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Figure 2.- Engine chamber pressure.

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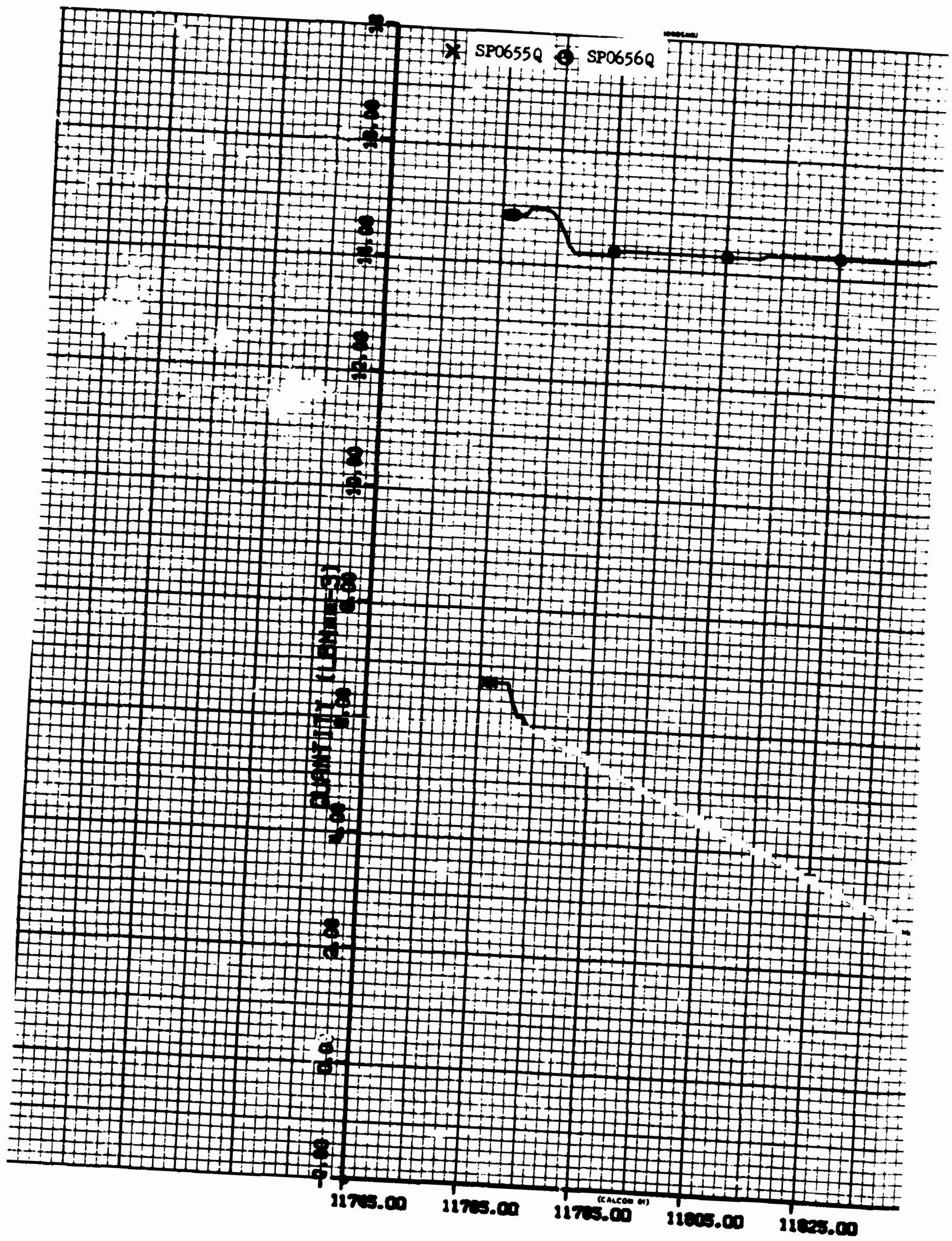

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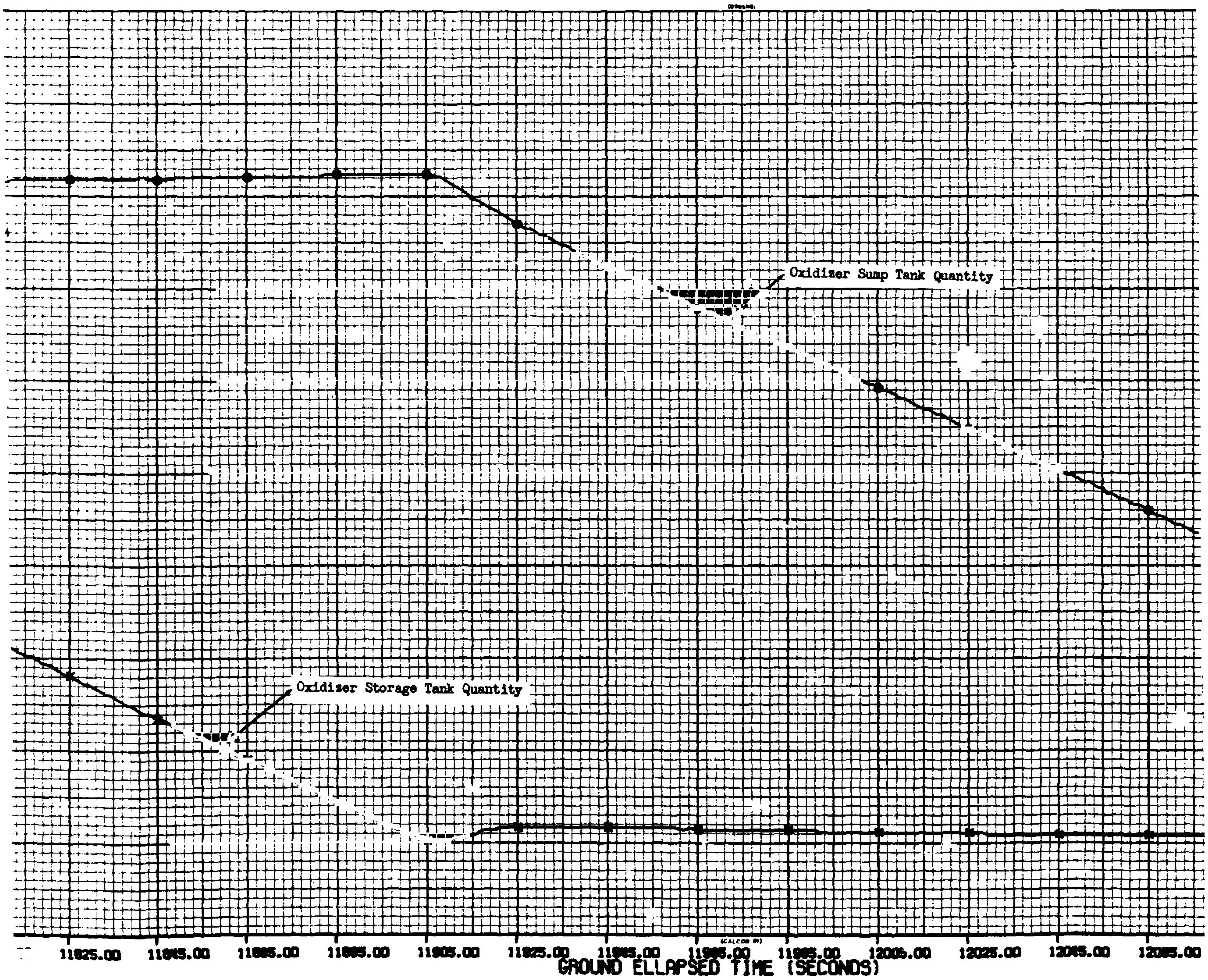

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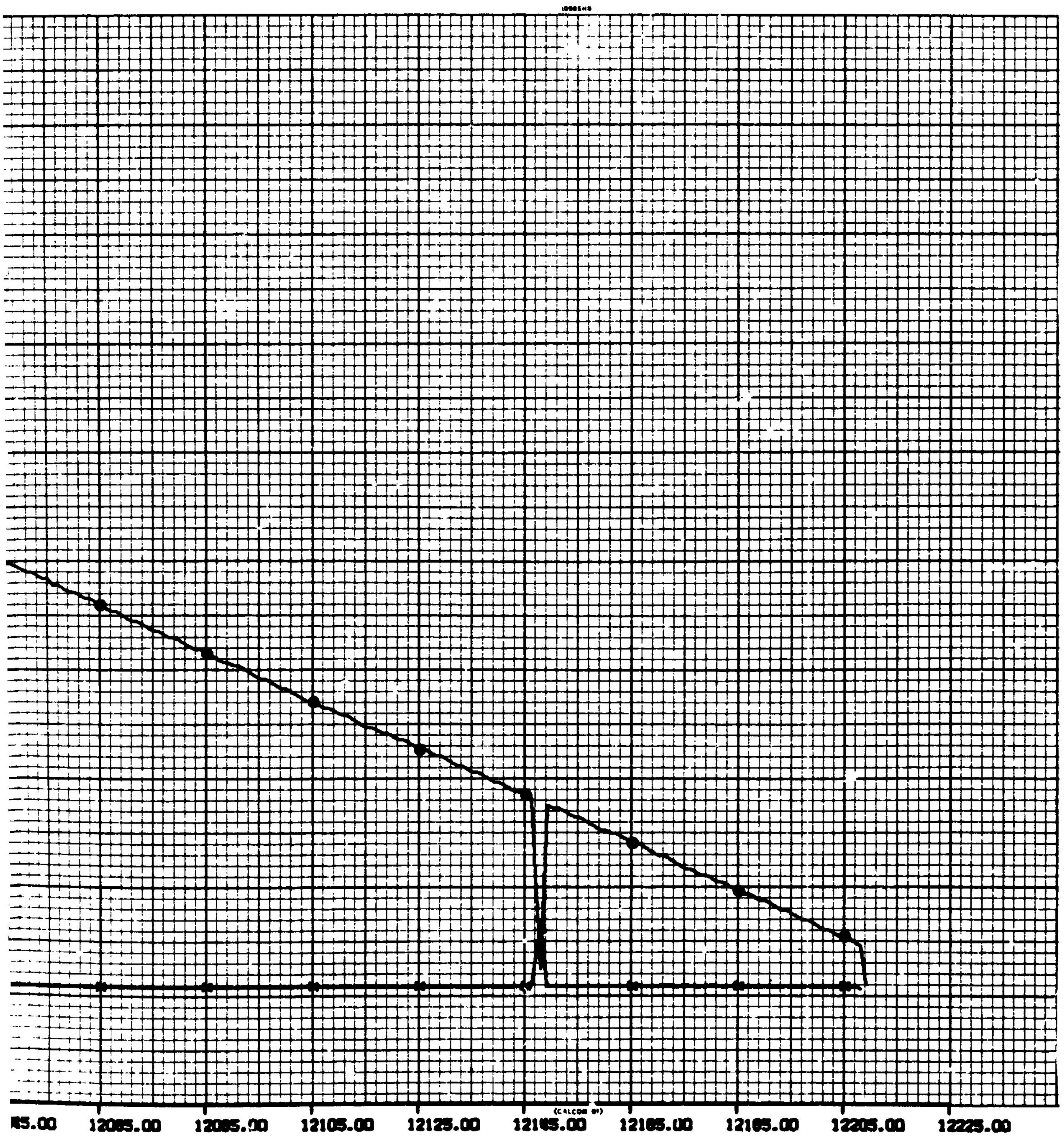

Figure 3.- Oxidizer primary gaging quantities.

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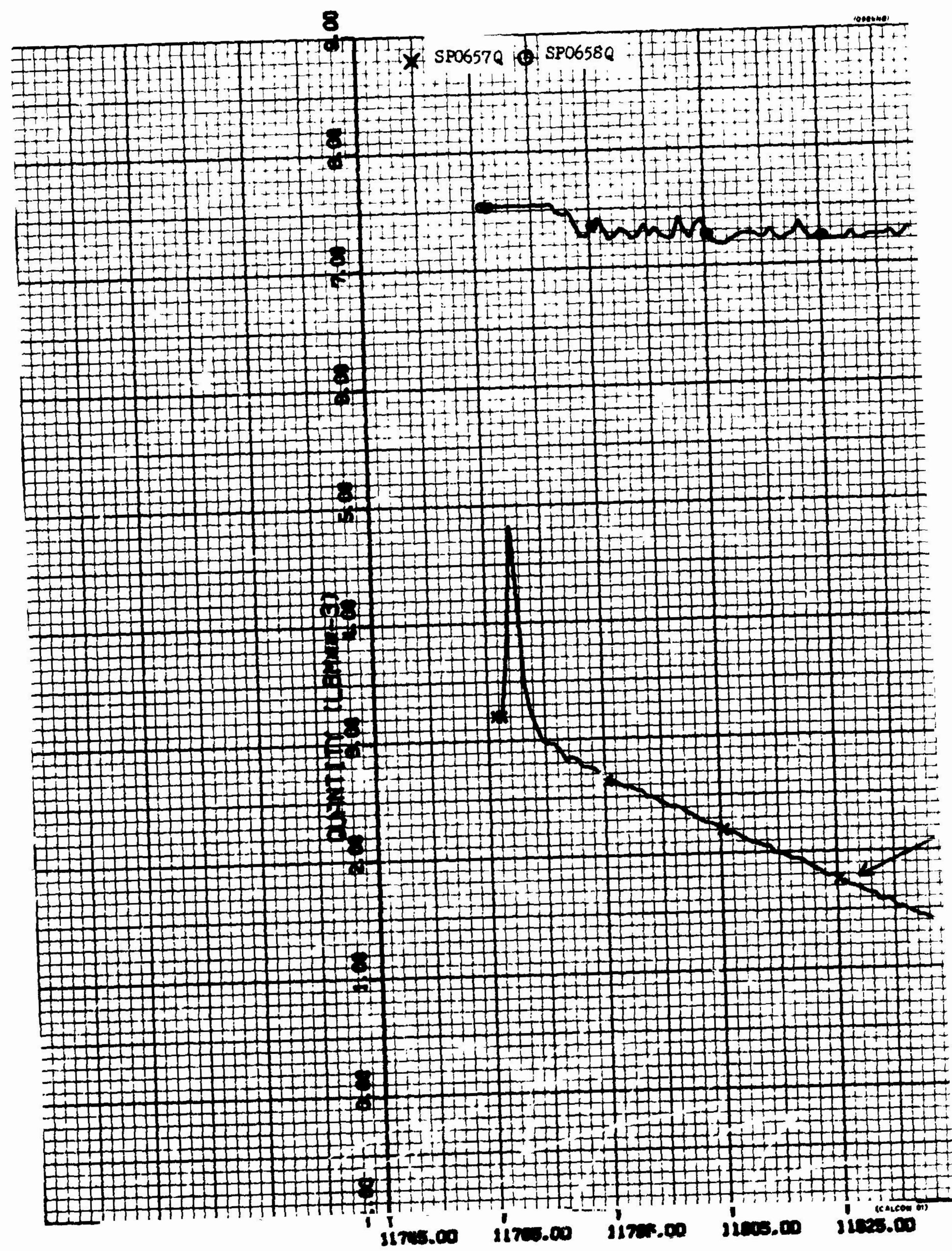

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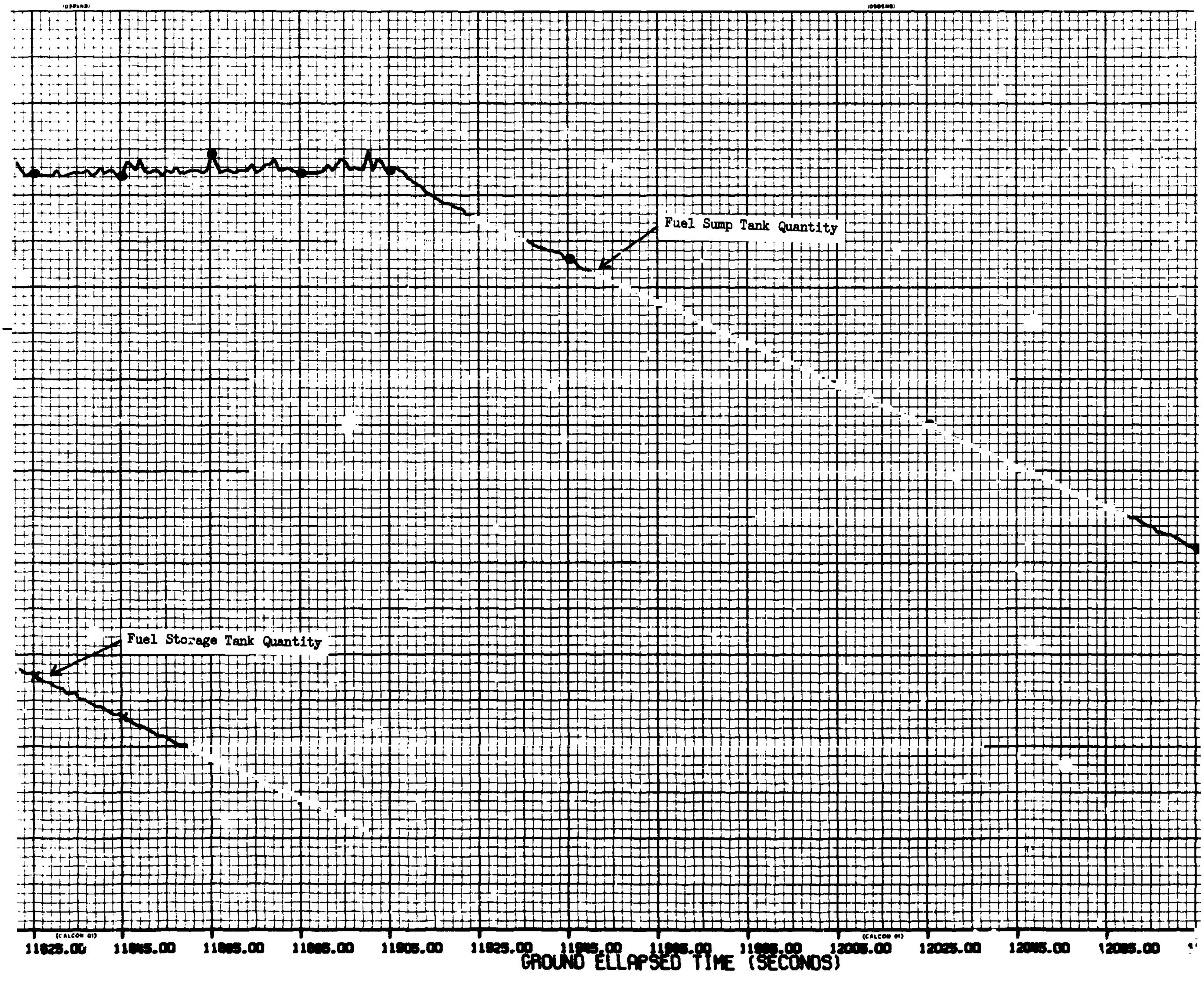

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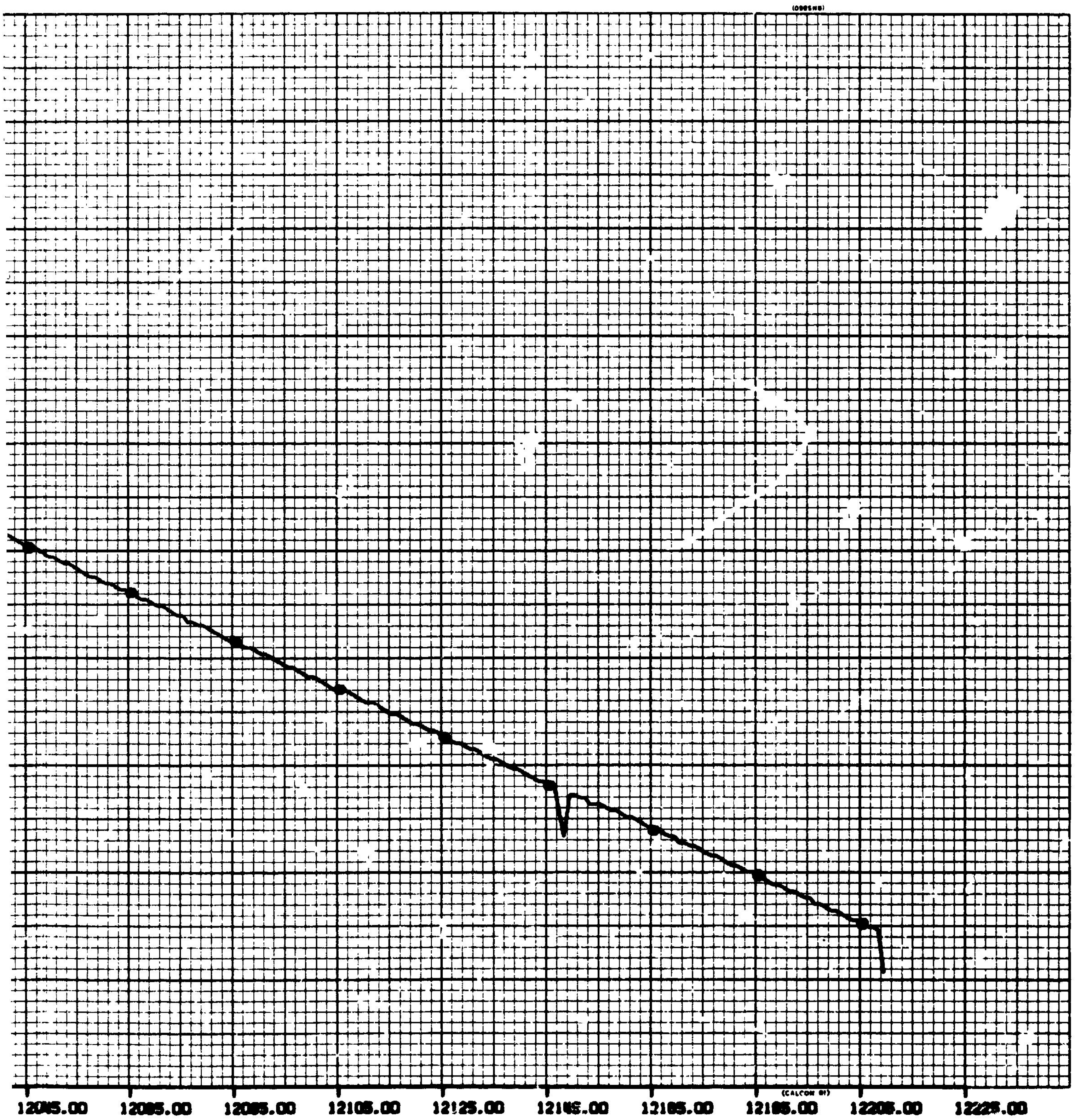

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Figure 4.- Fuel primary gaging quantities.

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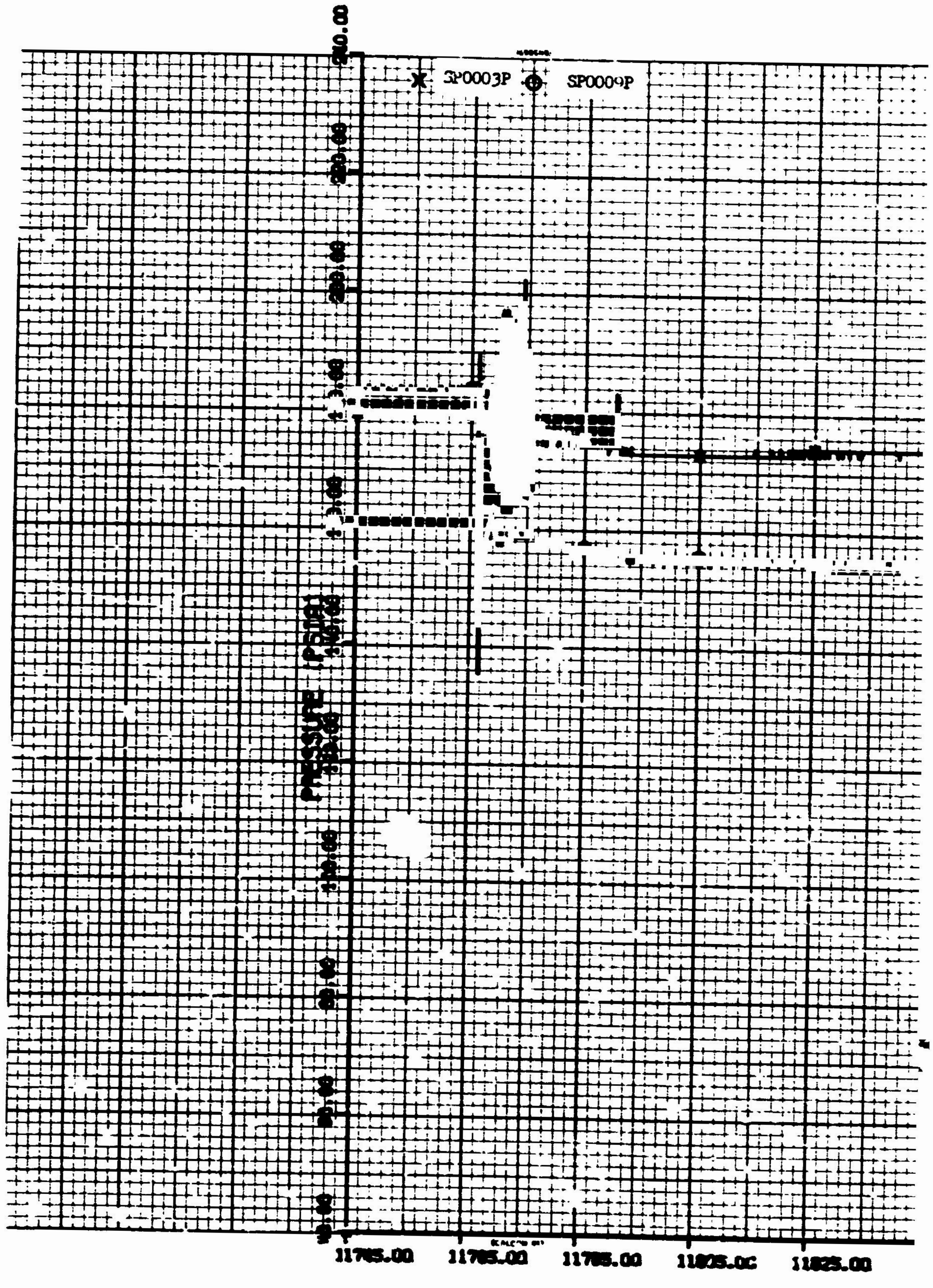

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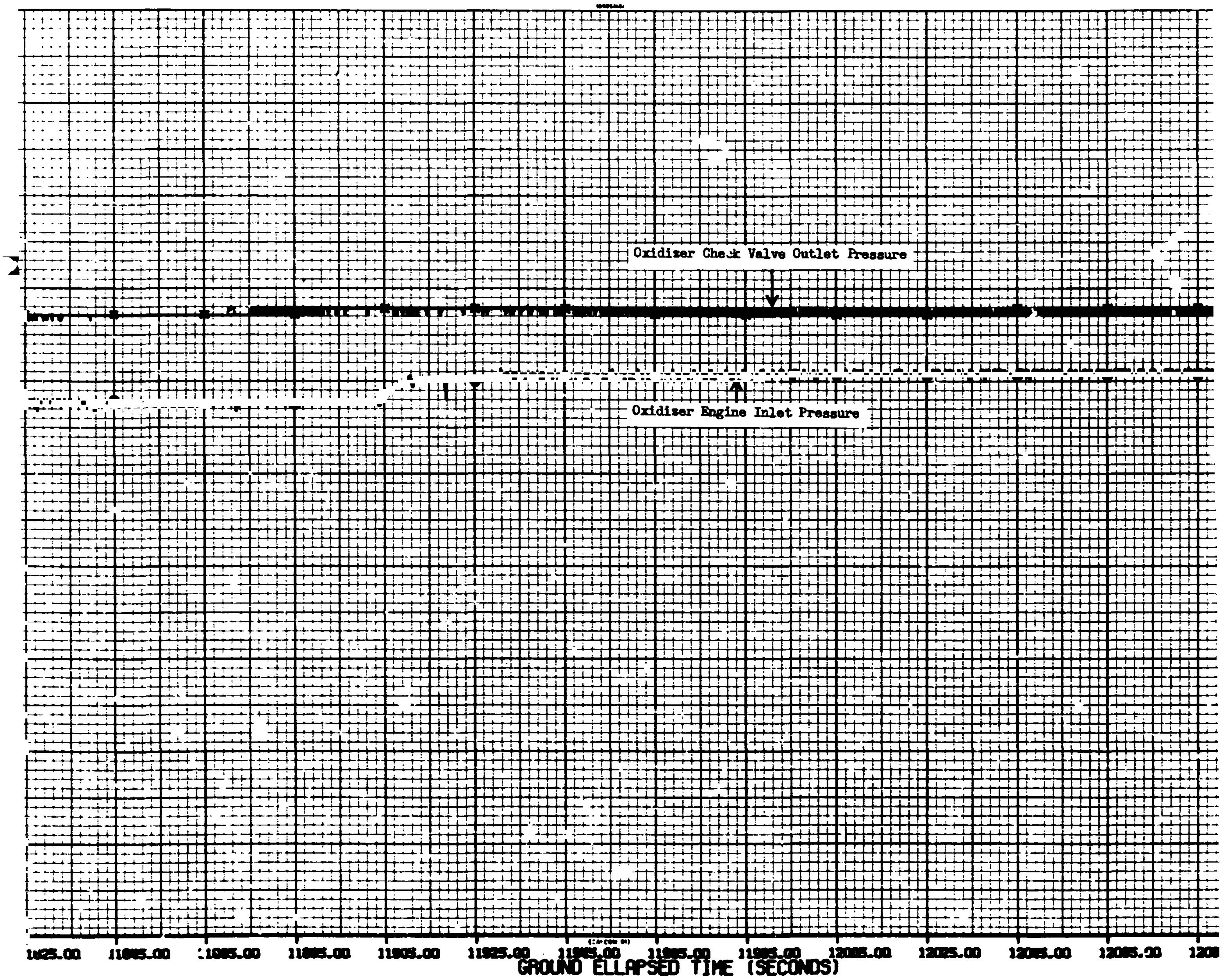

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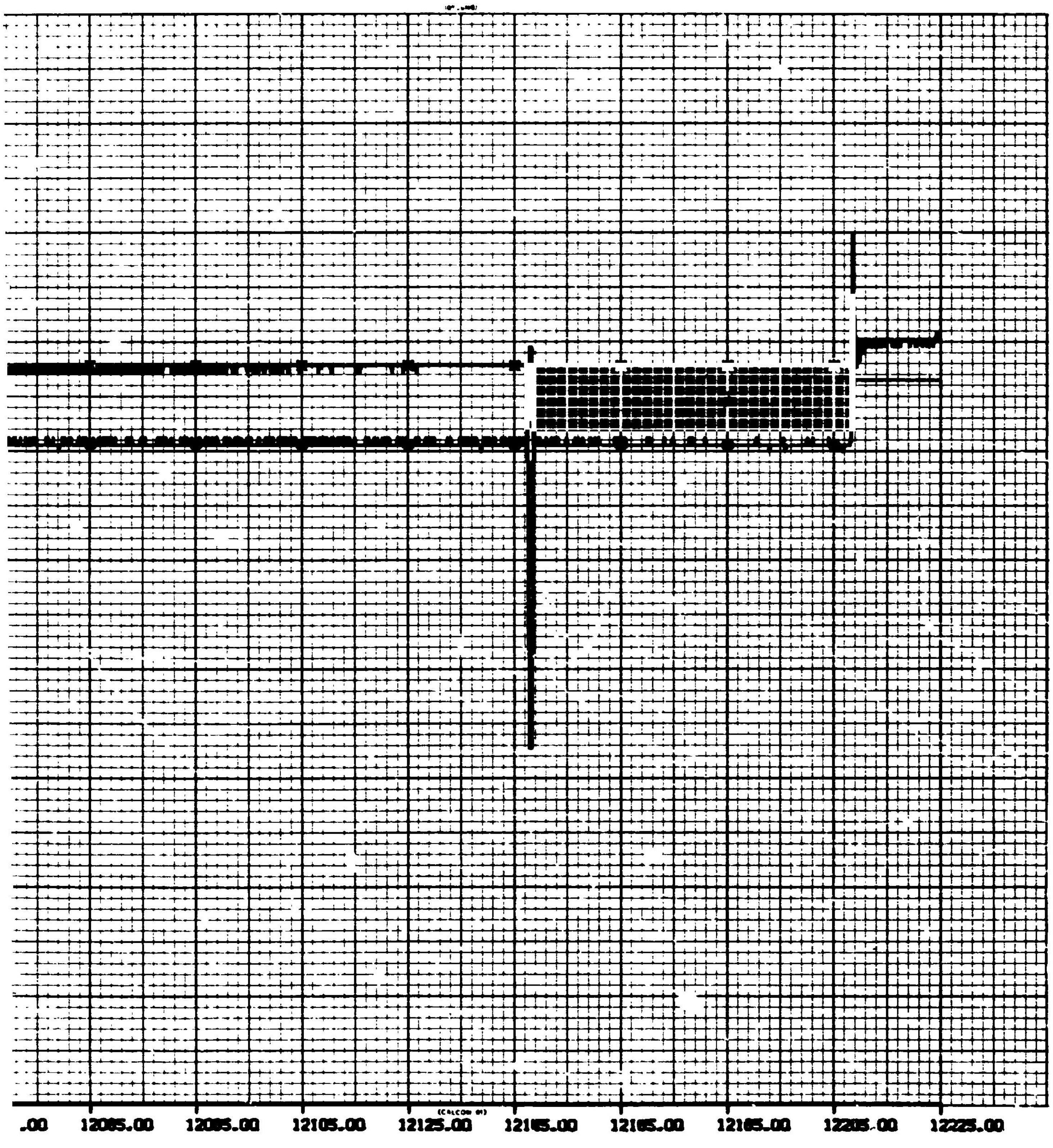

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Figure 5.- Oxidizer system pressures.

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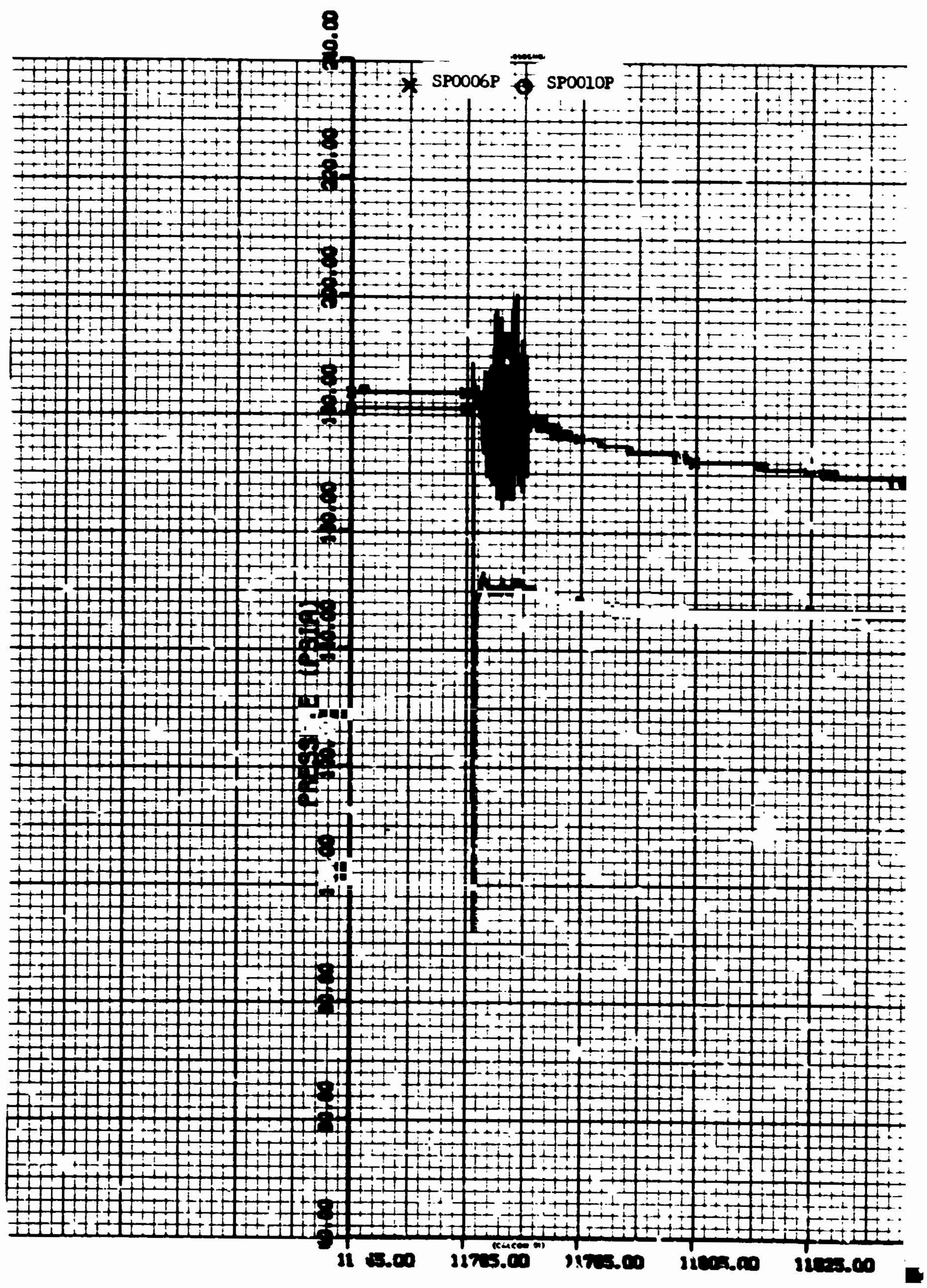

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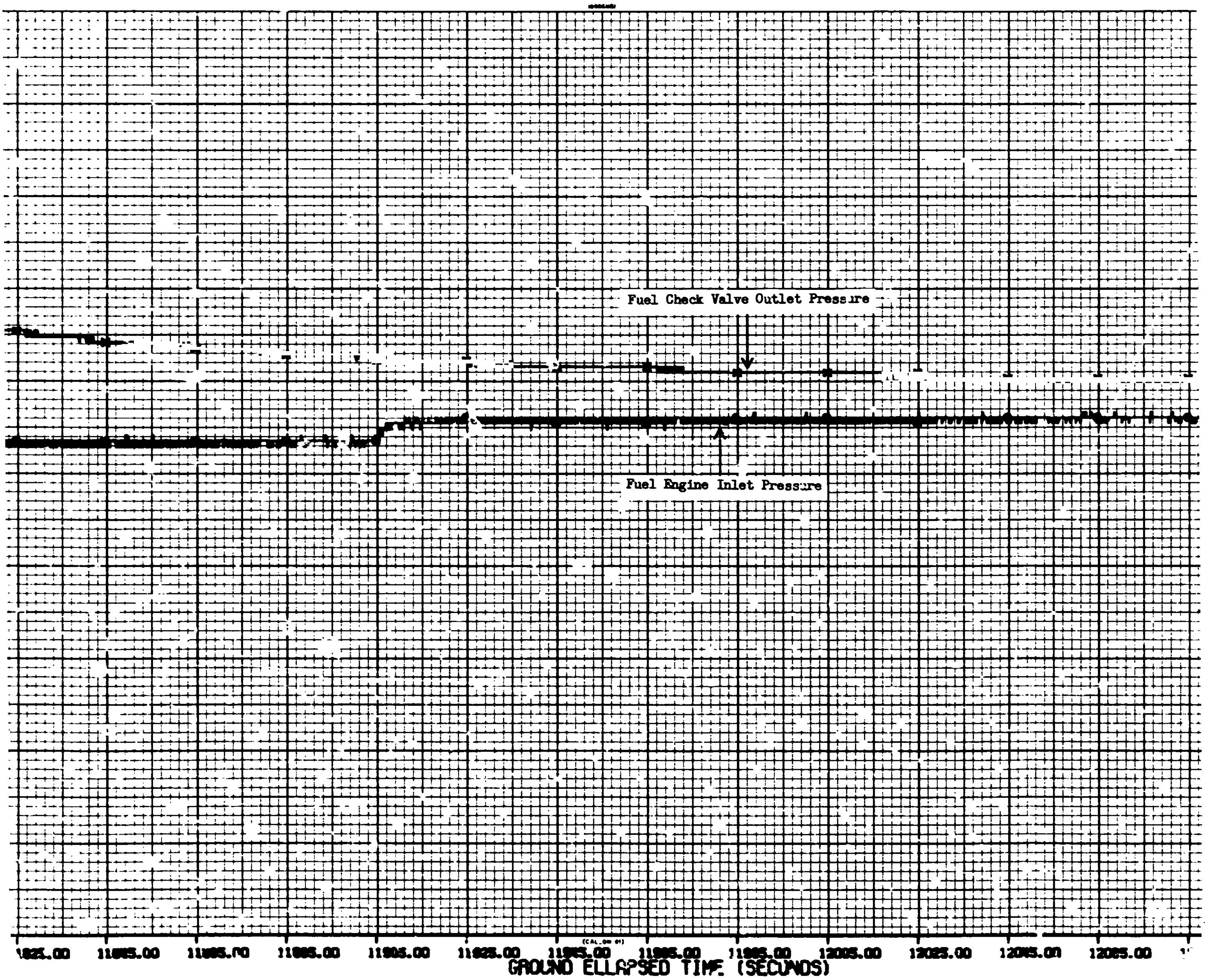

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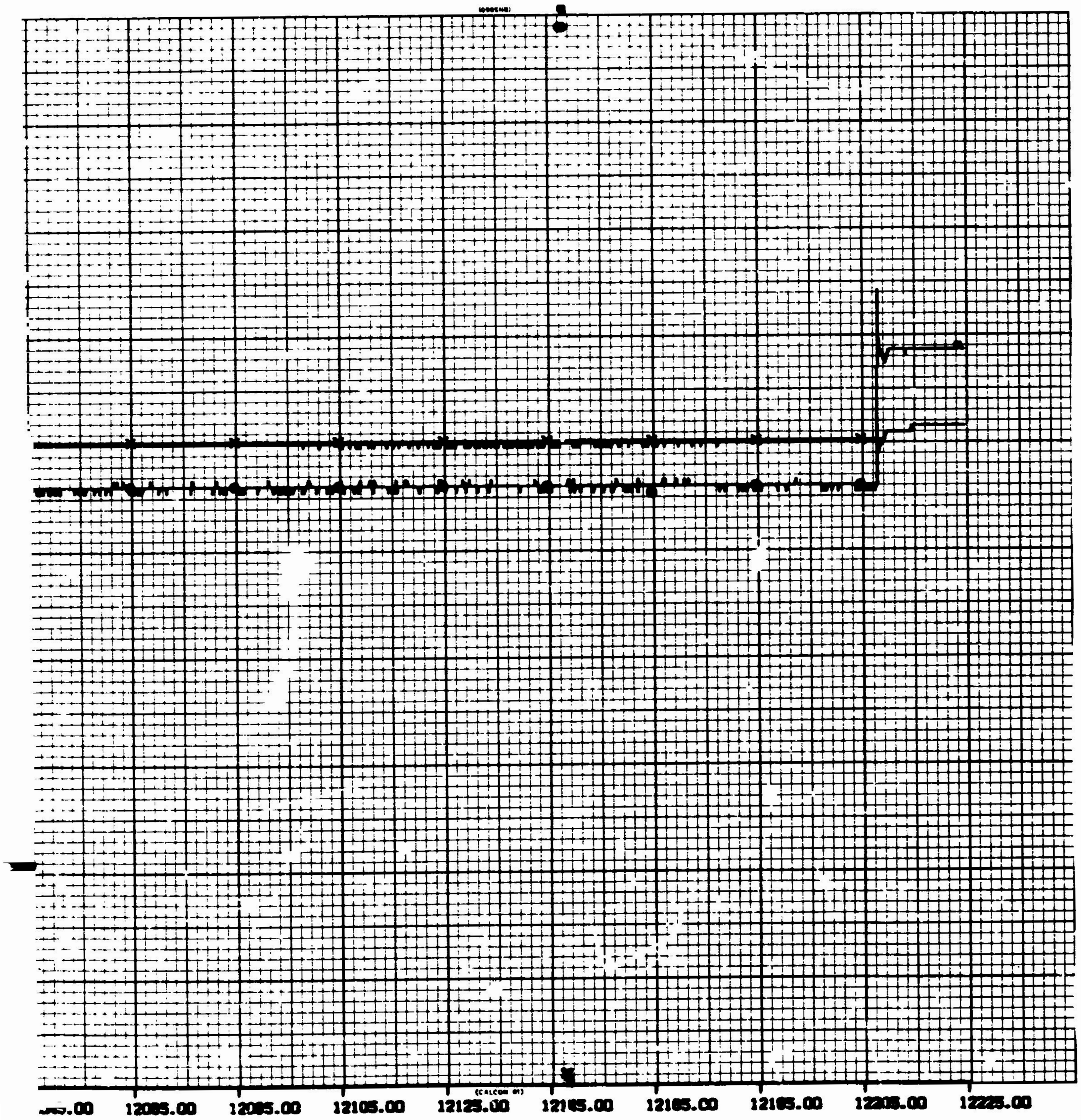

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Figure 6.- Fuel system pressures.

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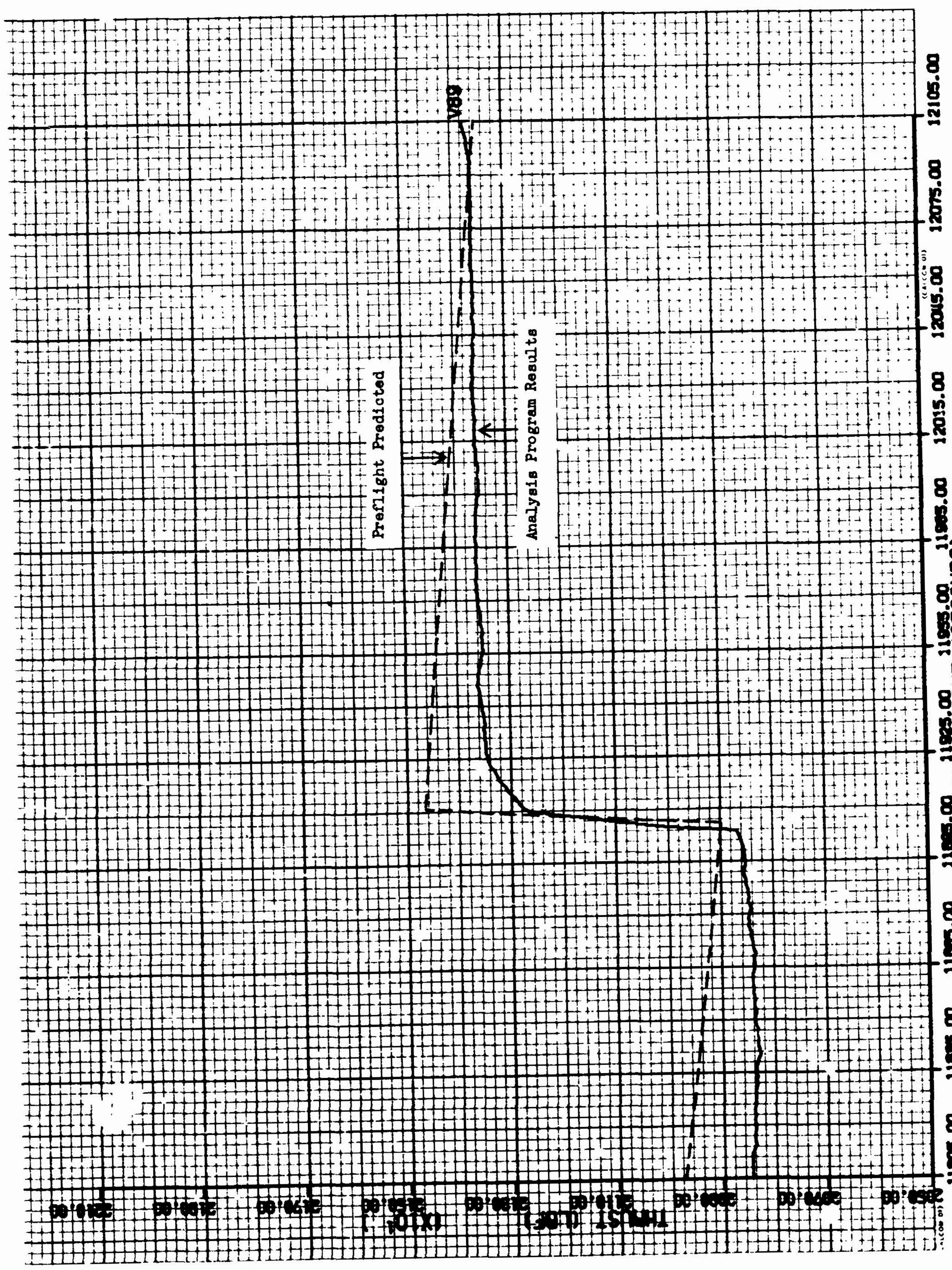


Figure 7.- Vacuum thrust.

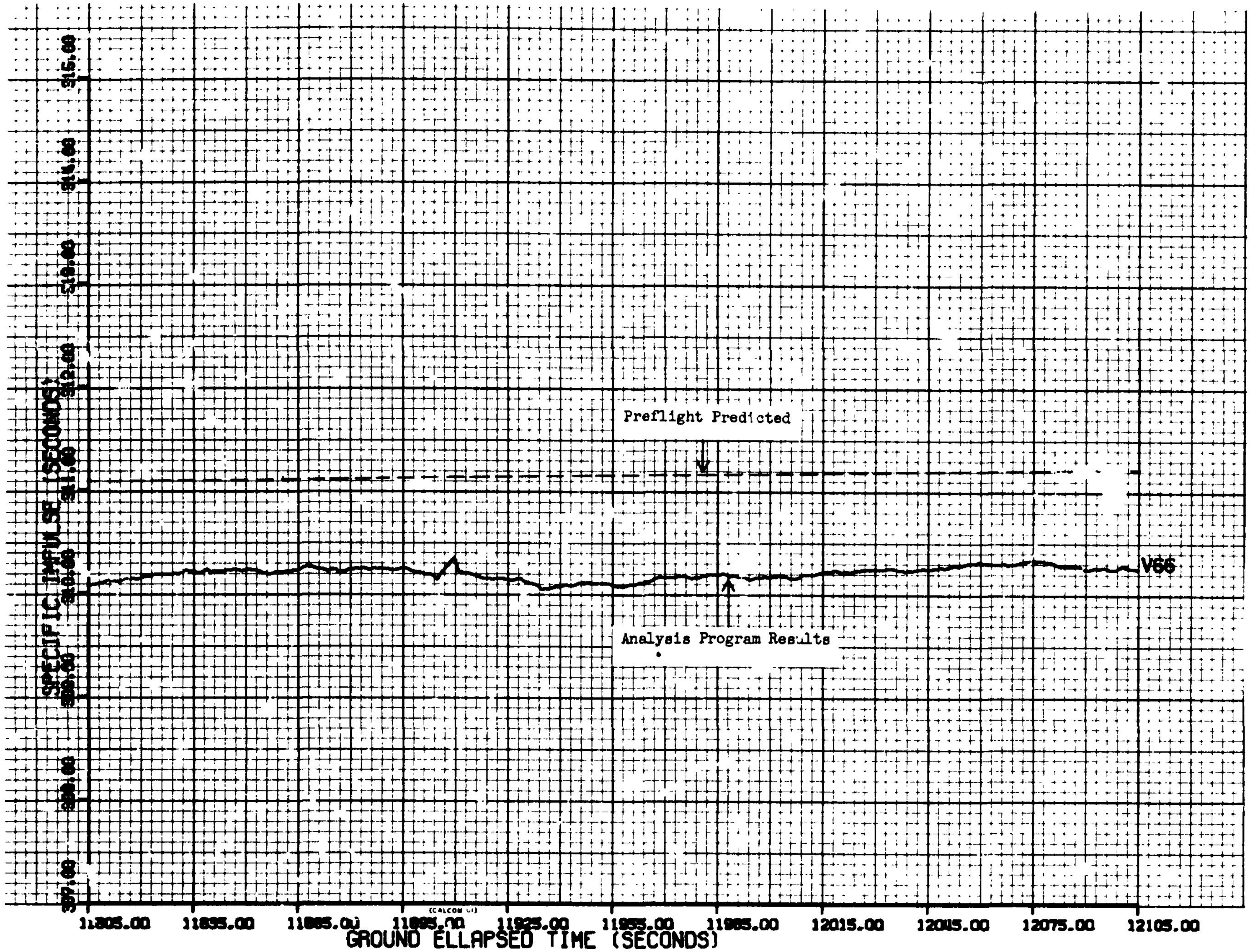


Figure 8.- Vacuum specific impulse.

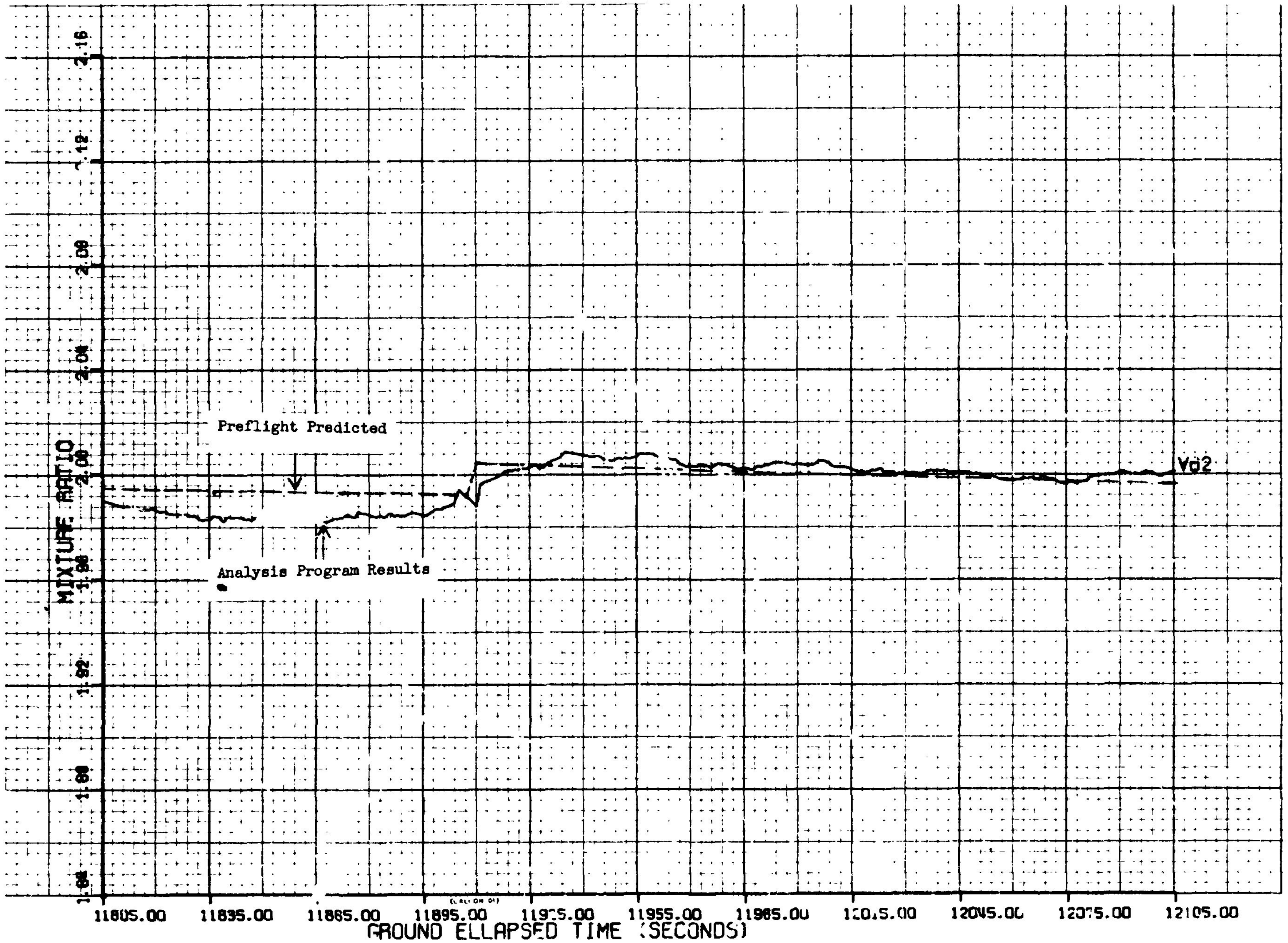


Figure 9.- Propellant mixture ratio.

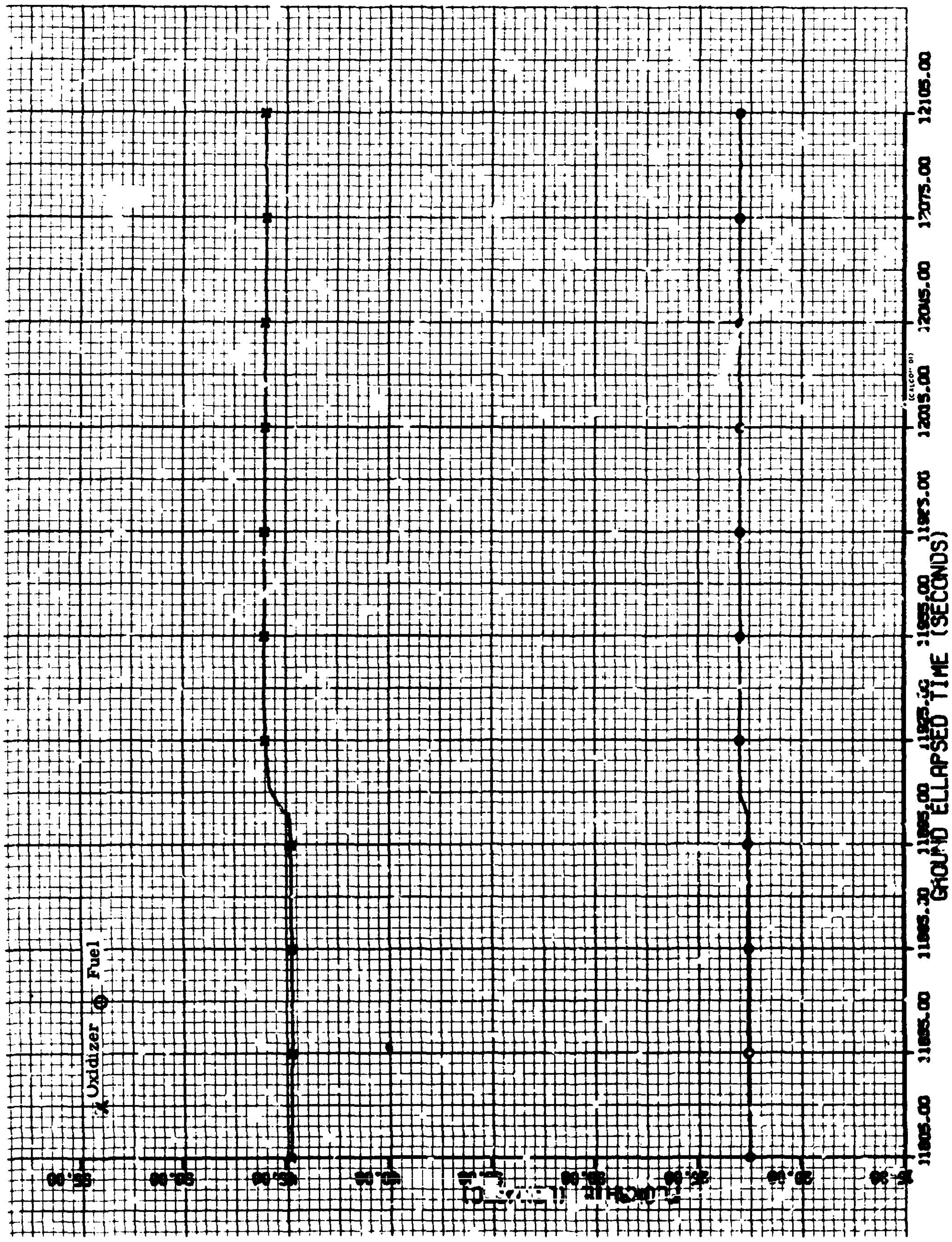


Figure 10.- Propellant flow rates.

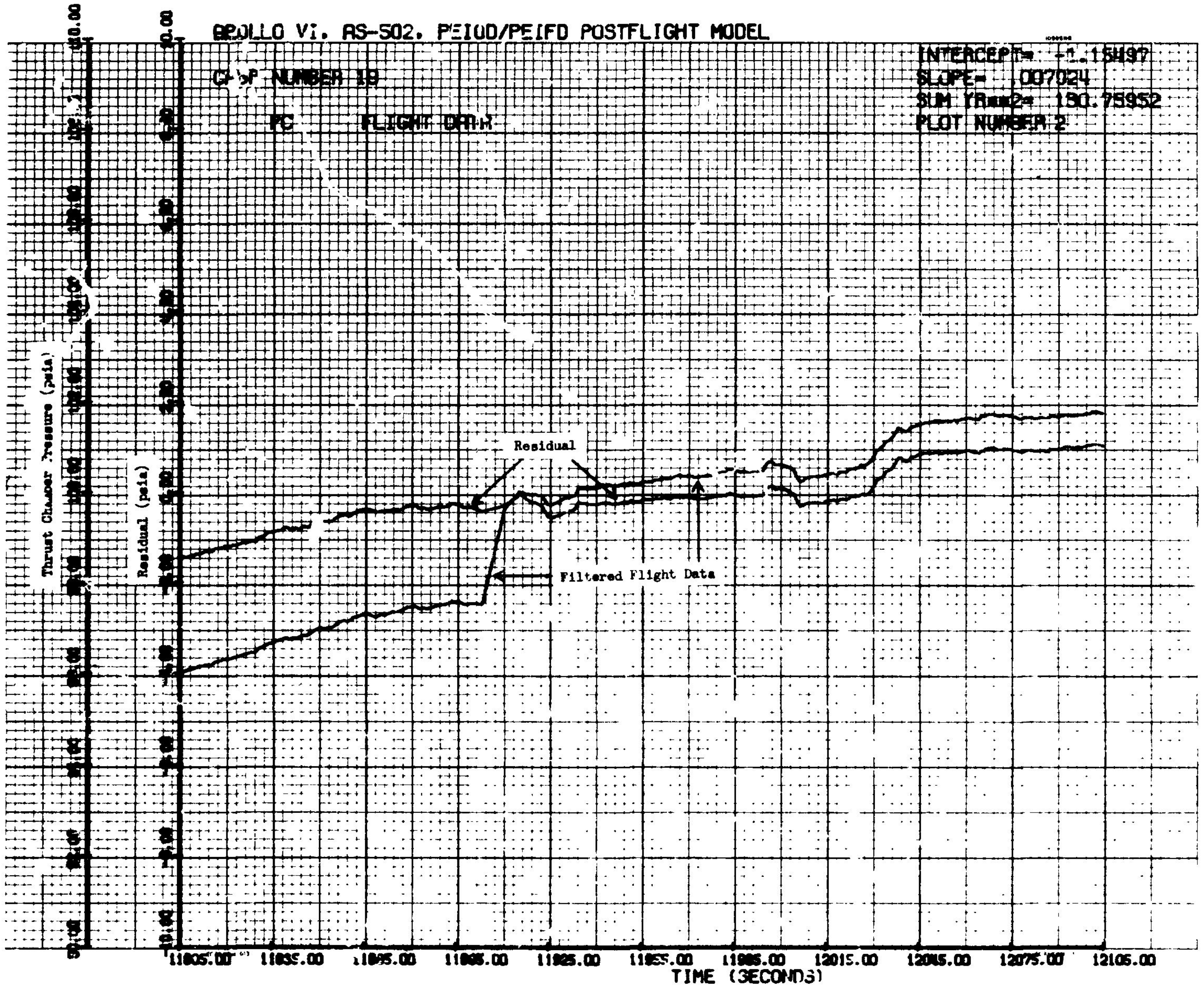


Figure 11. - Engine chamber pressure match.

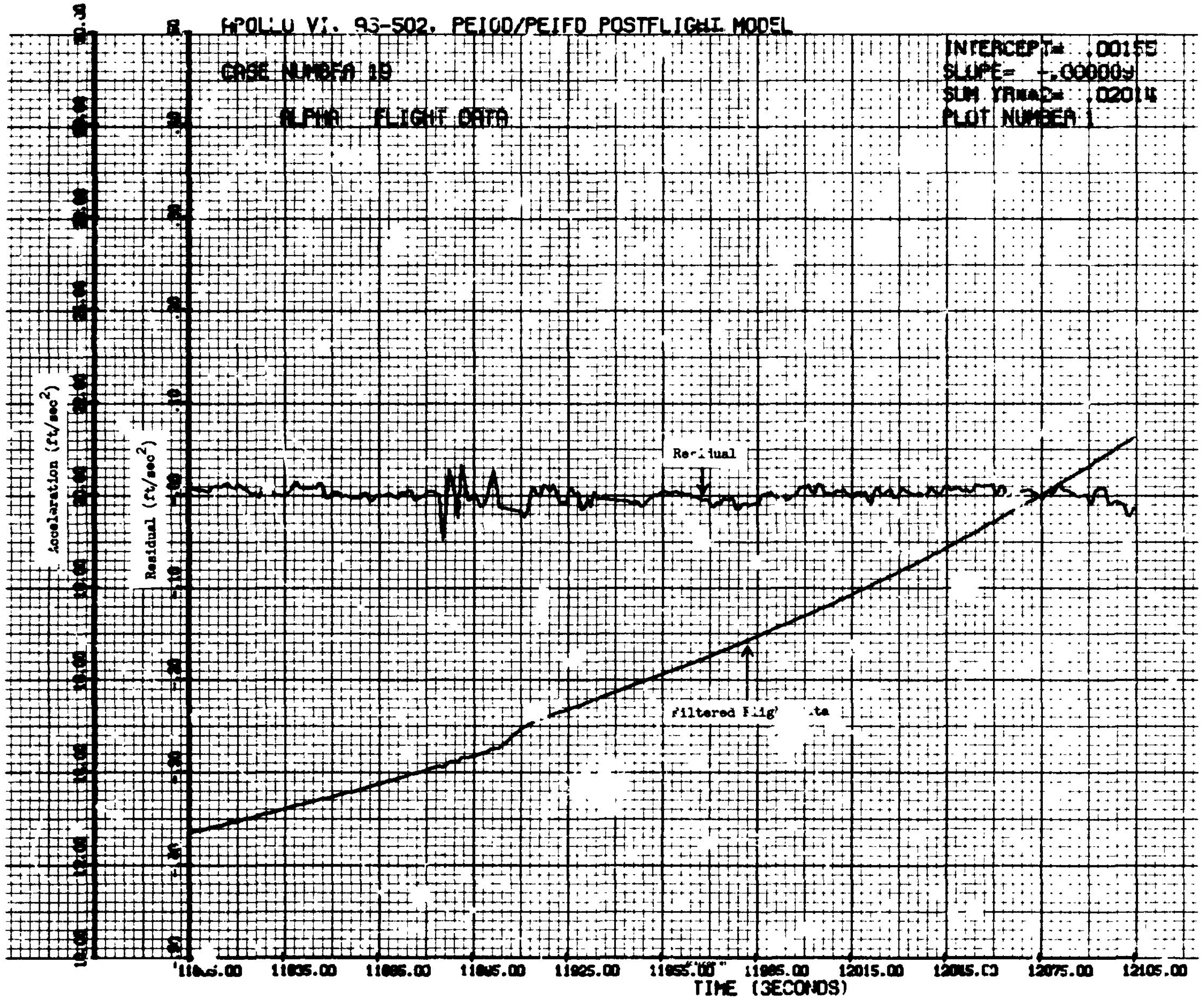


Figure 12.- Acceleration match.

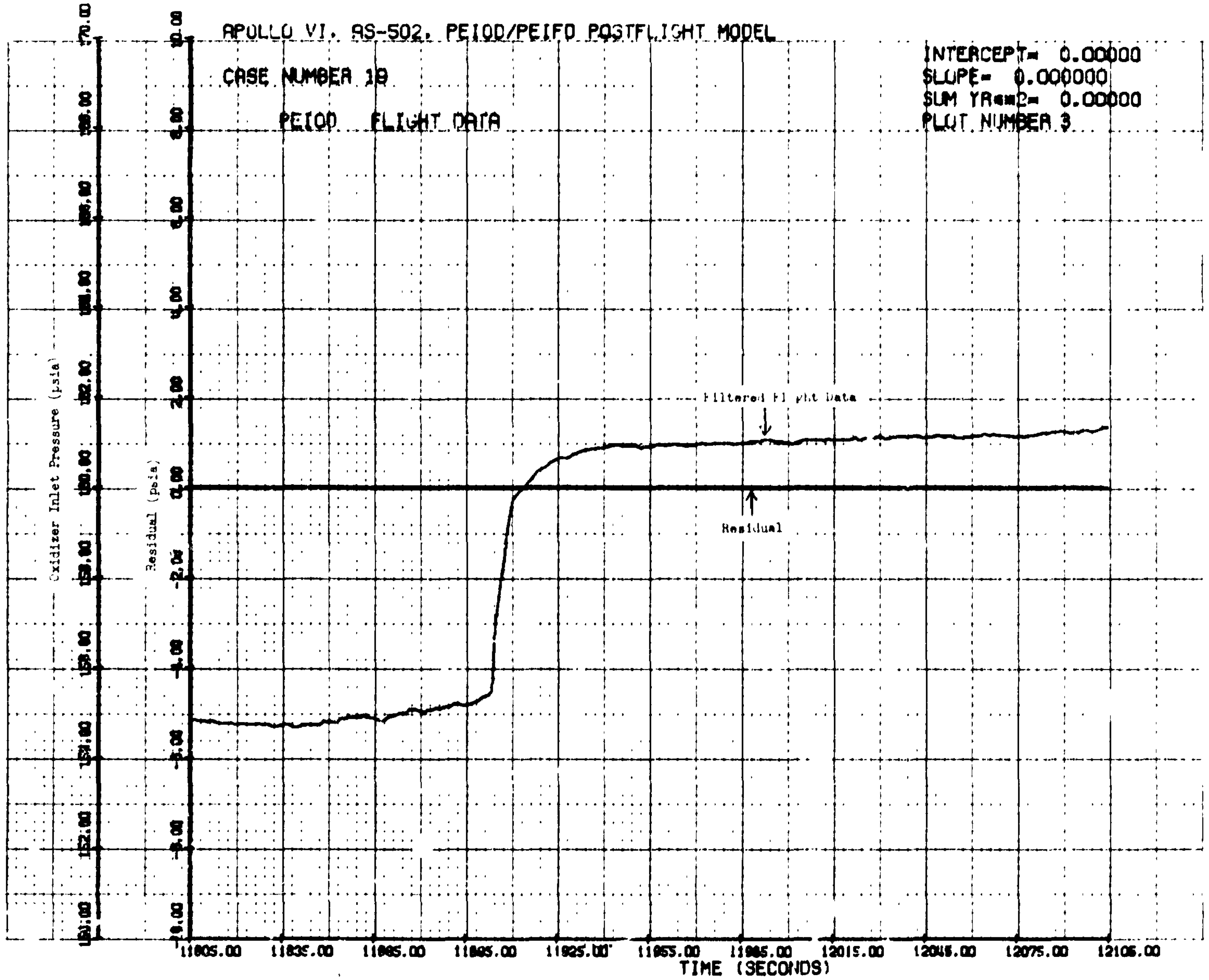


Figure 13. - Oxidizer inlet pressure match.

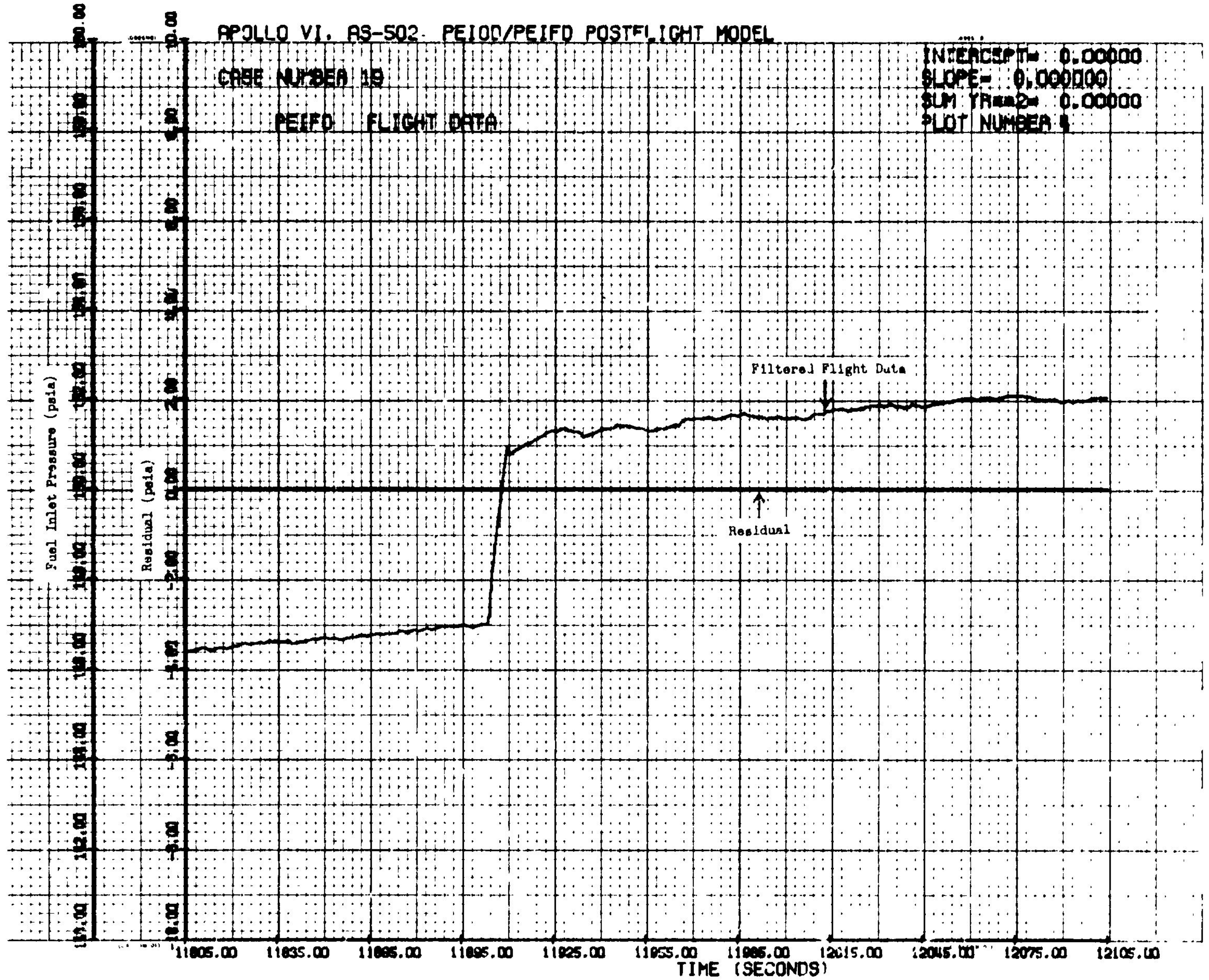


Figure 14. - Fuel inlet pressure match.

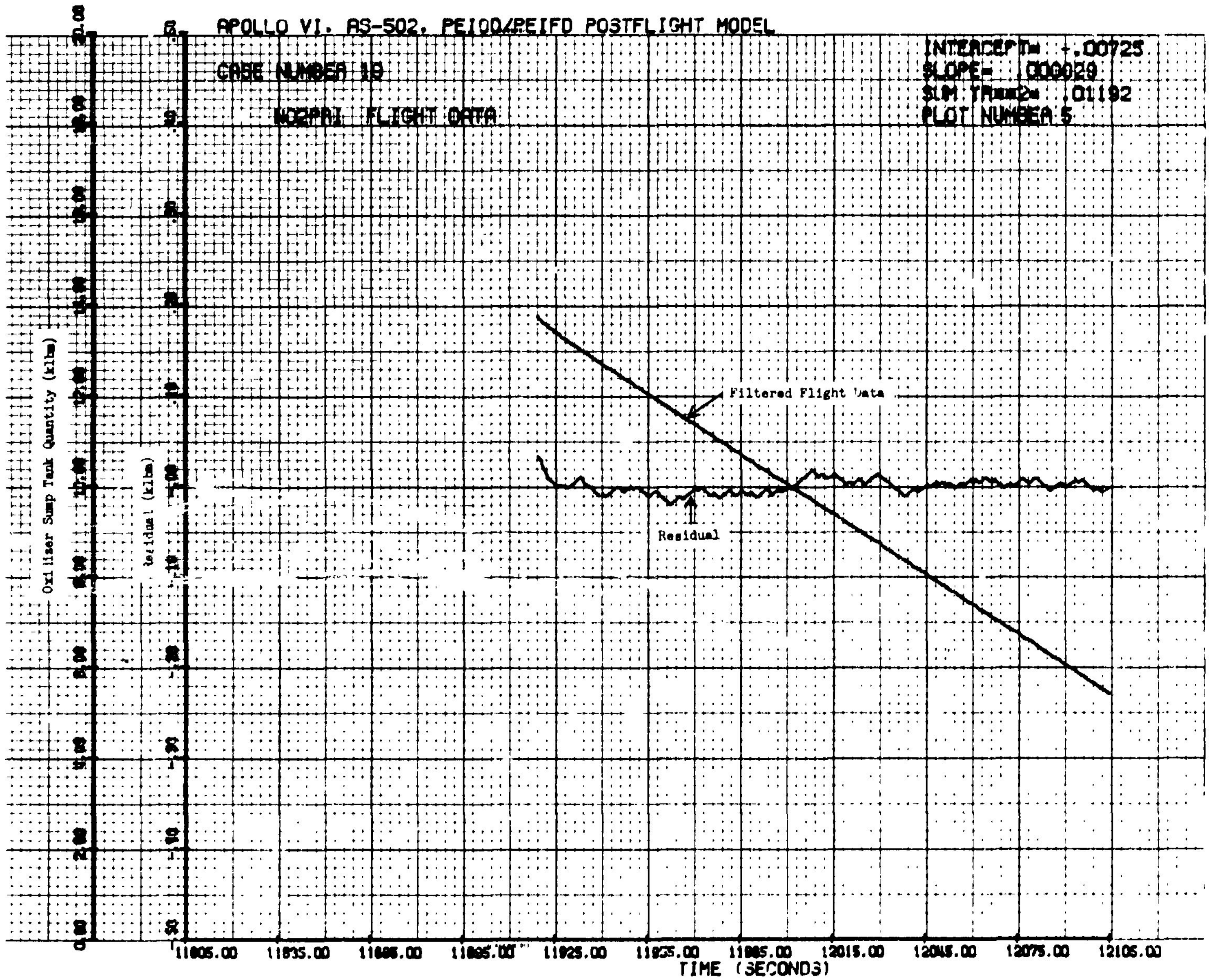


Figure 15. - Oxidizer sump tank primary gaging quantity match.

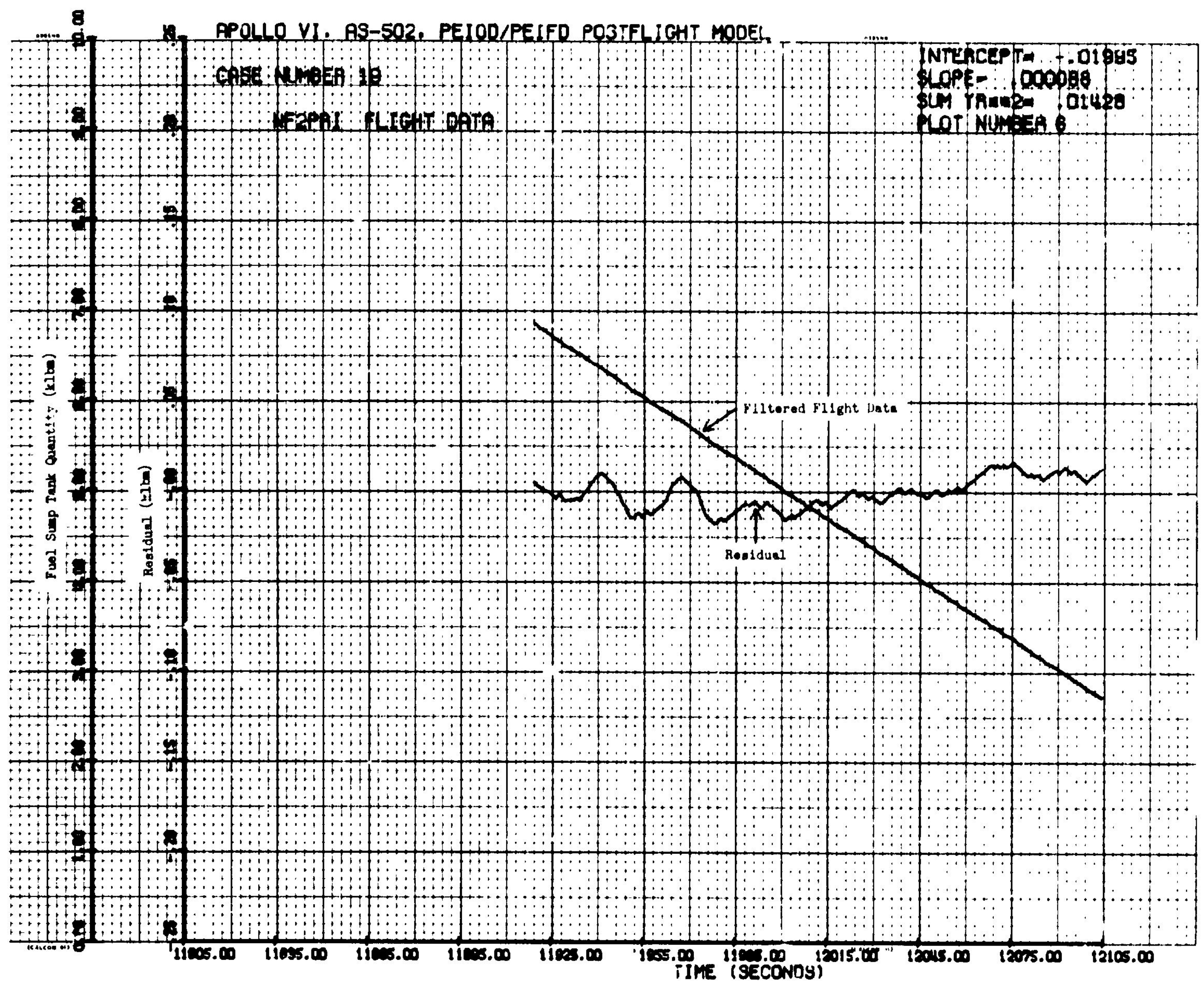


Figure 16.- Fuel sump tank primary gaging quantity match.

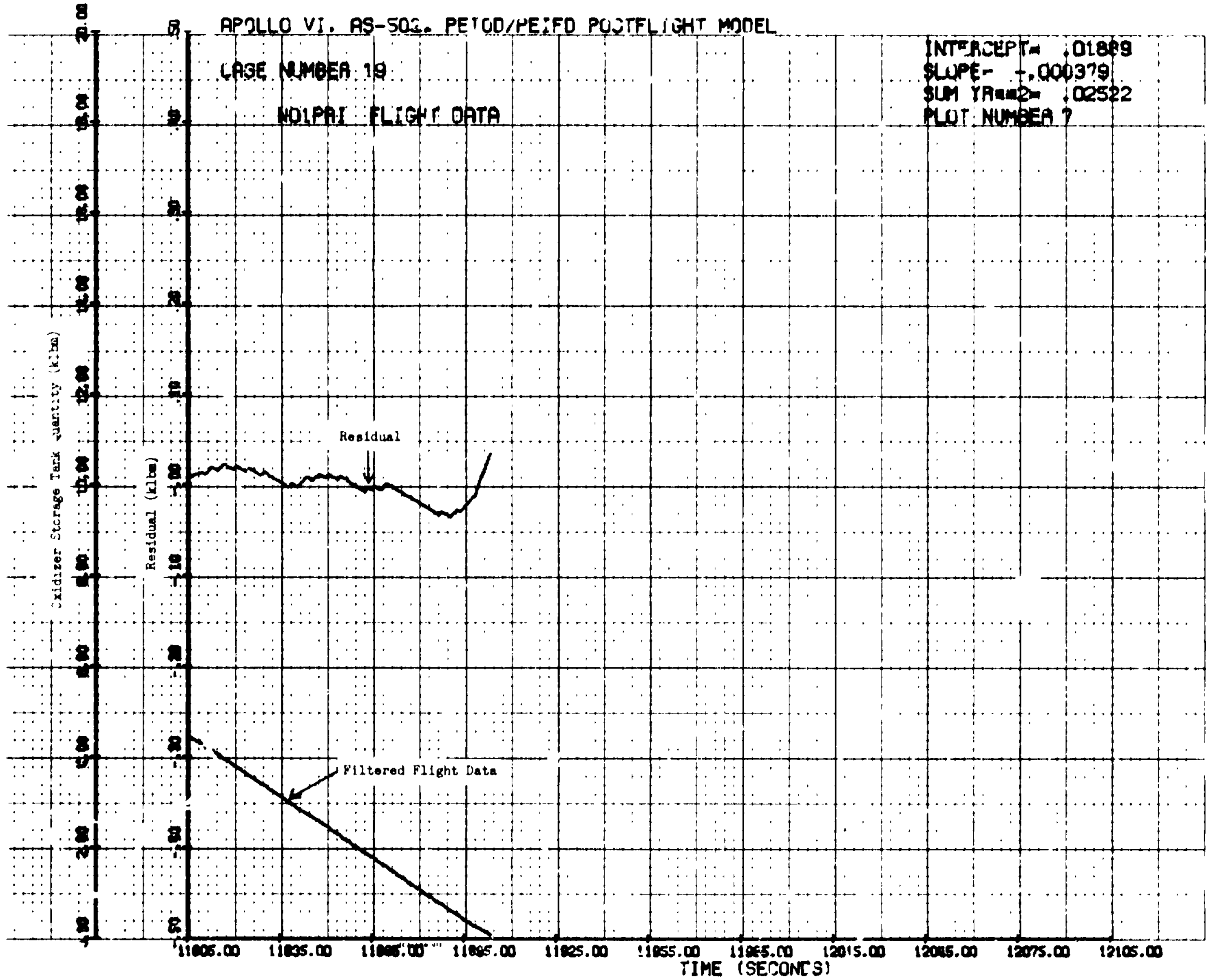


Figure 17. - Oxidizer storage tank primary gaging quantity match.

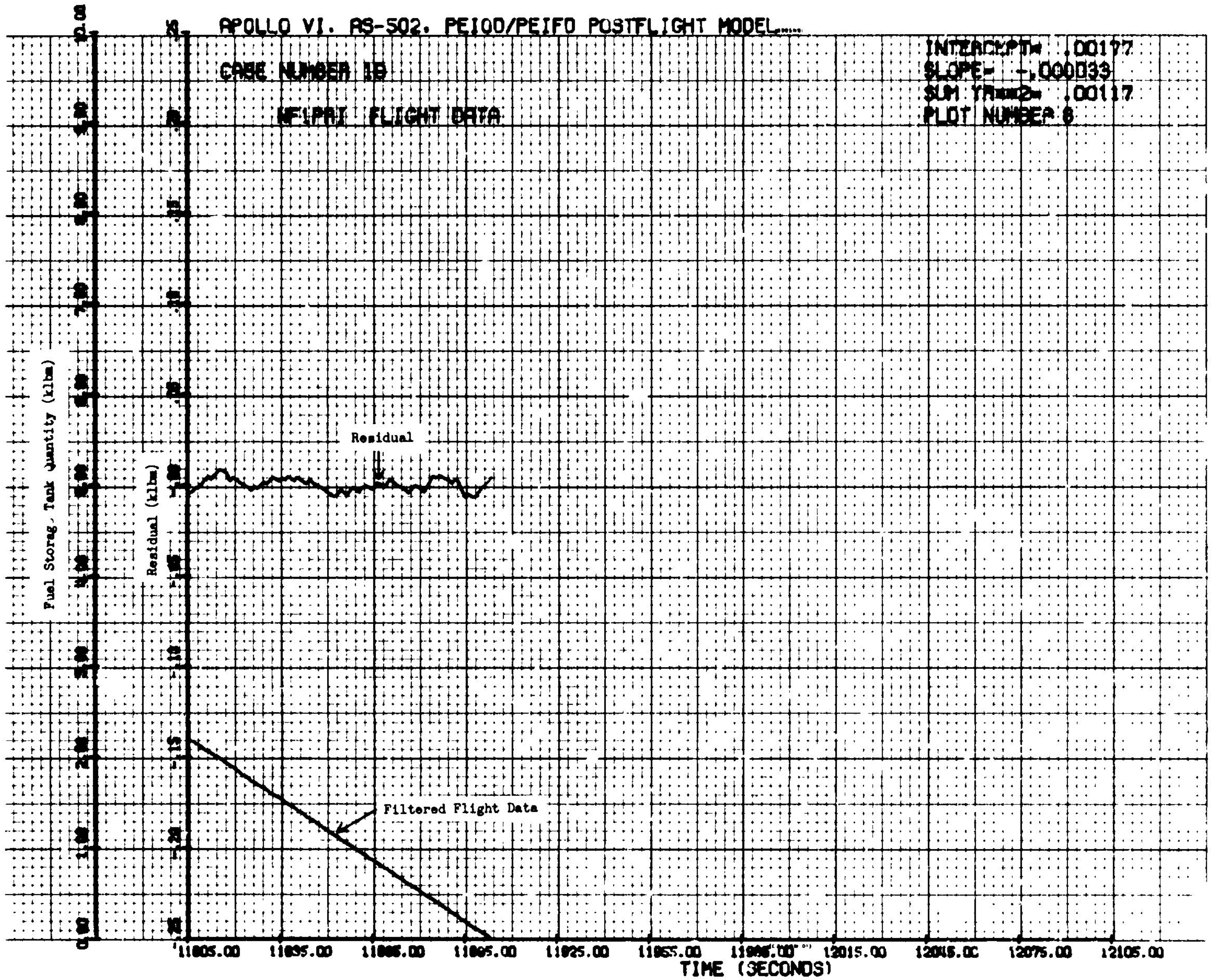
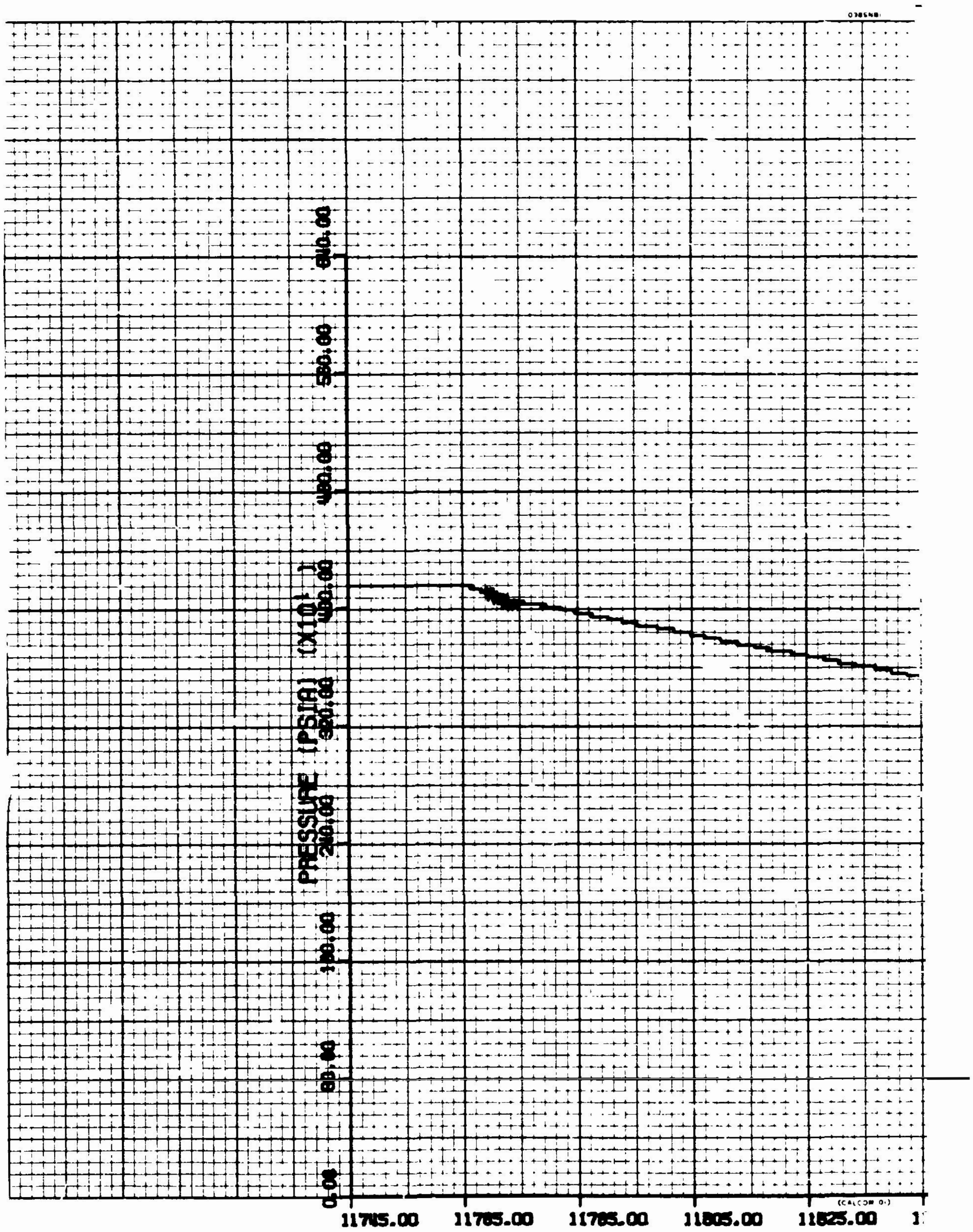


Figure 18. - Fuel st rage tank primary gaging quanti'v match.



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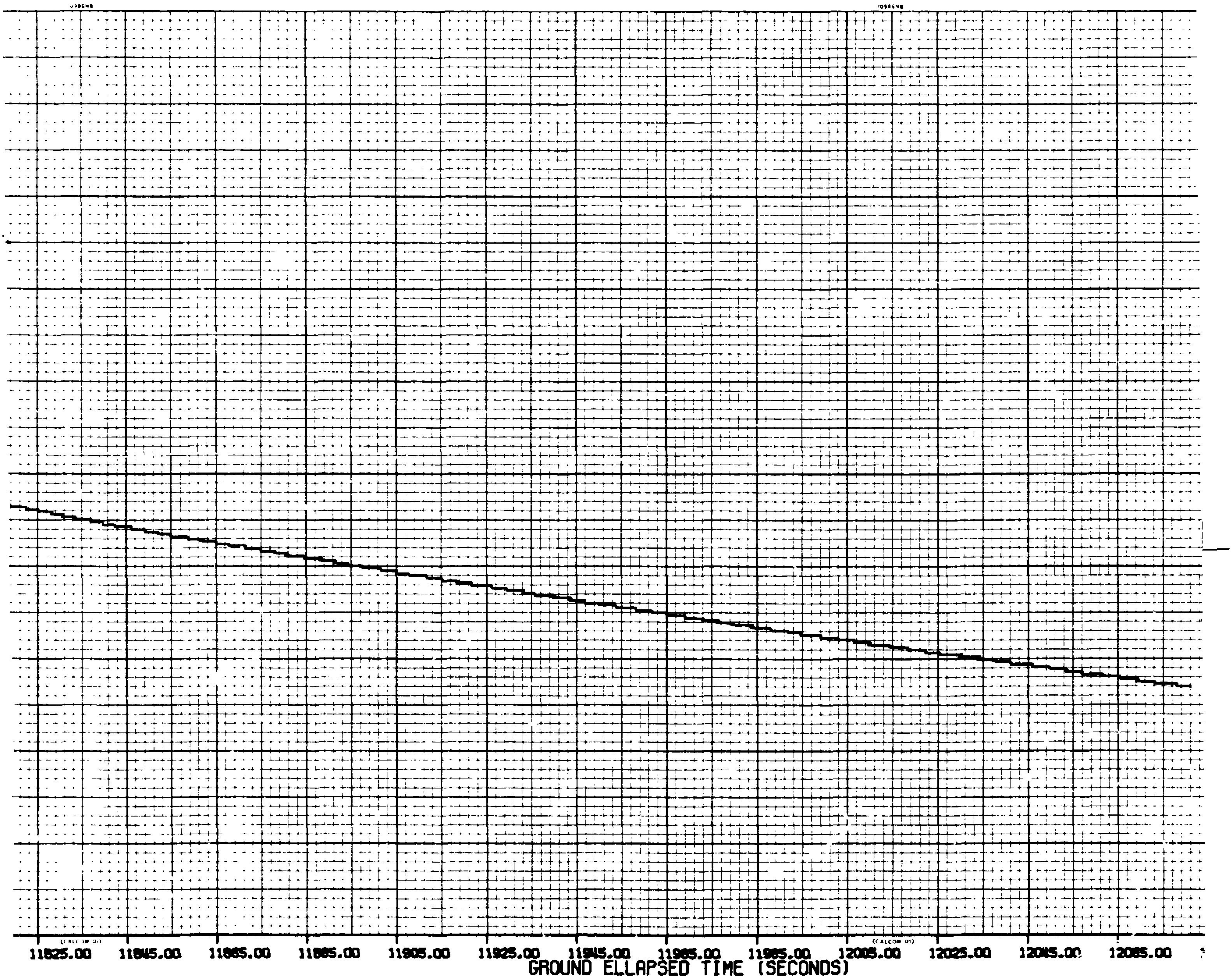

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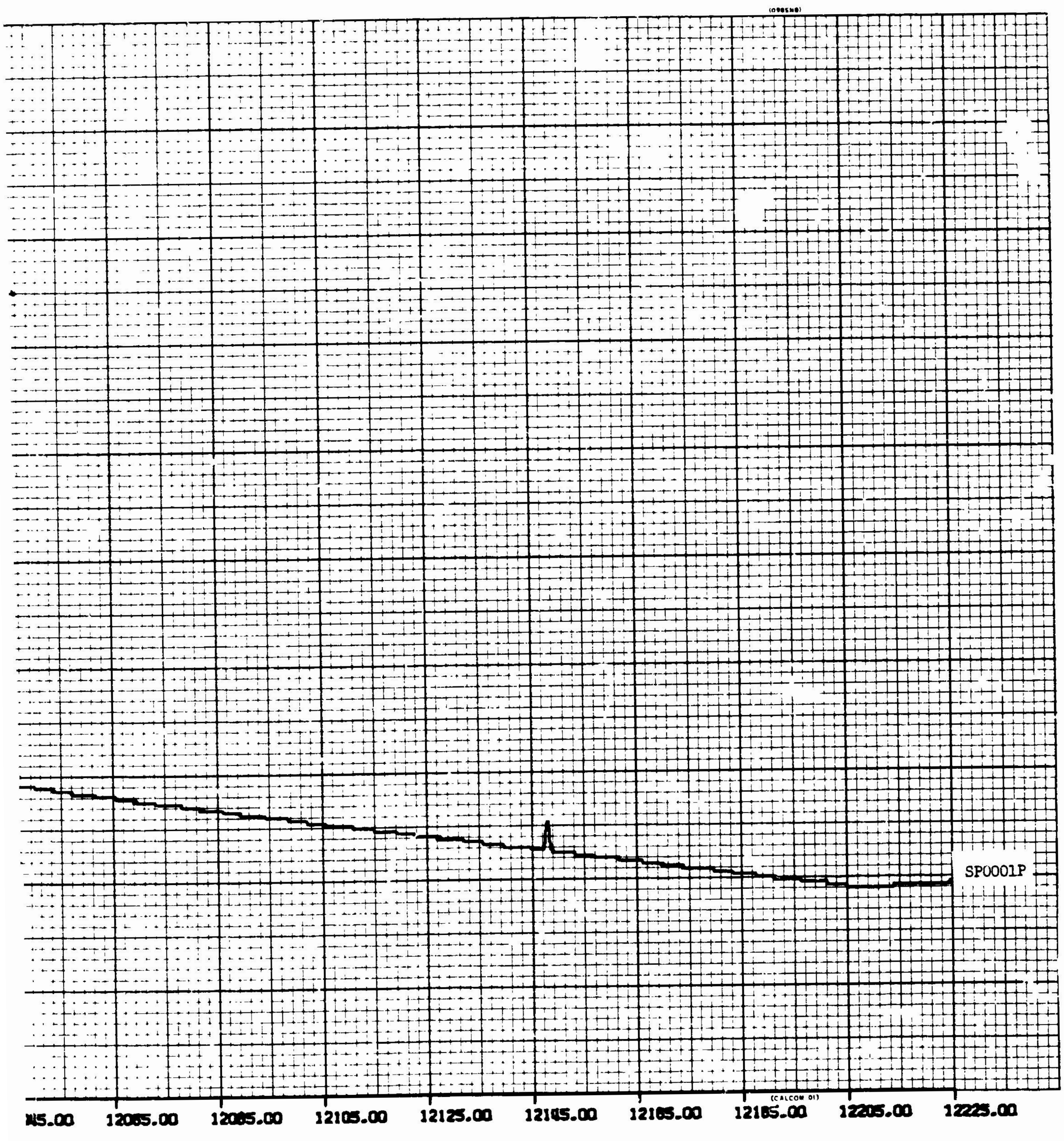

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Figure 19.- Helium bottle pressure.

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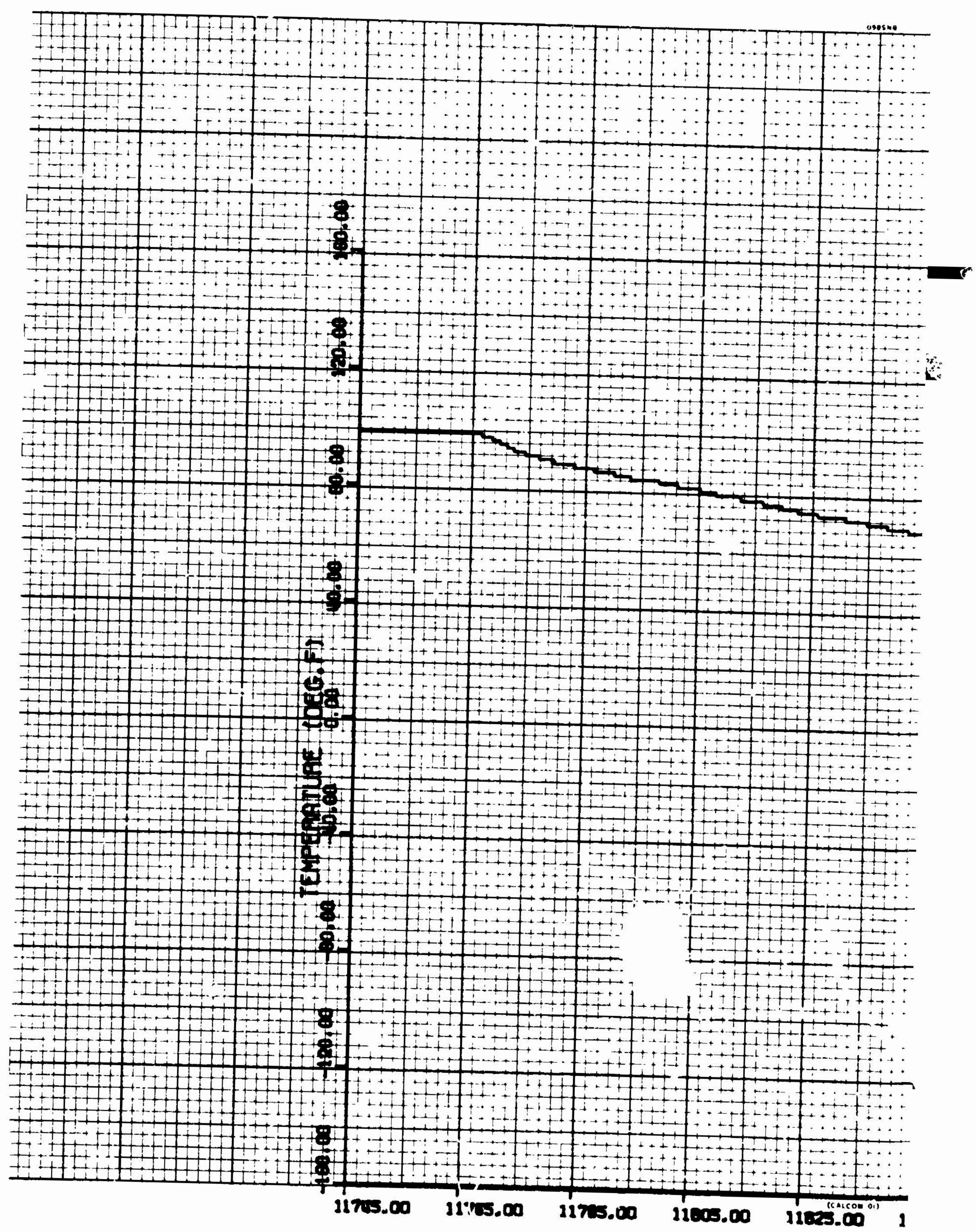

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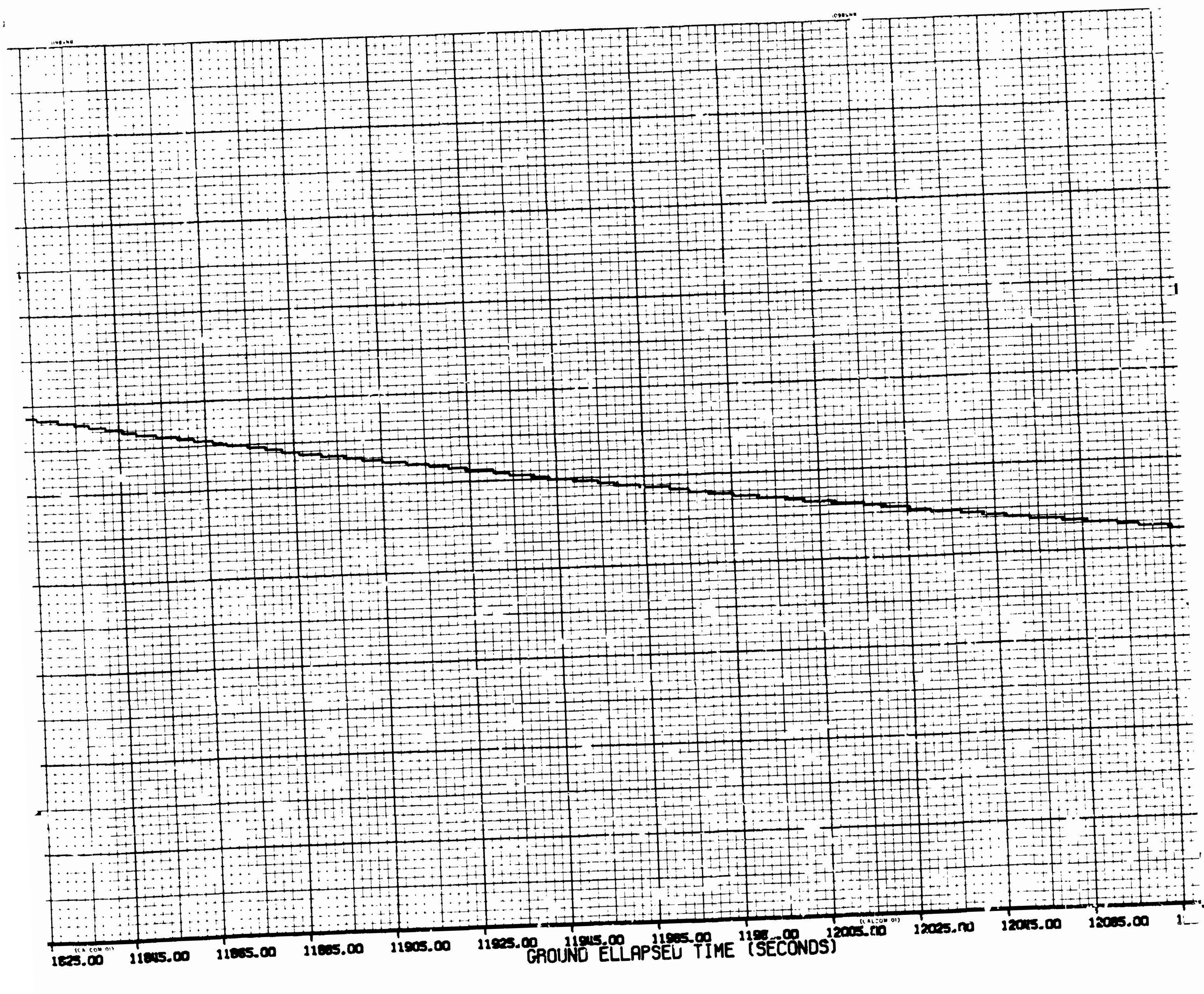

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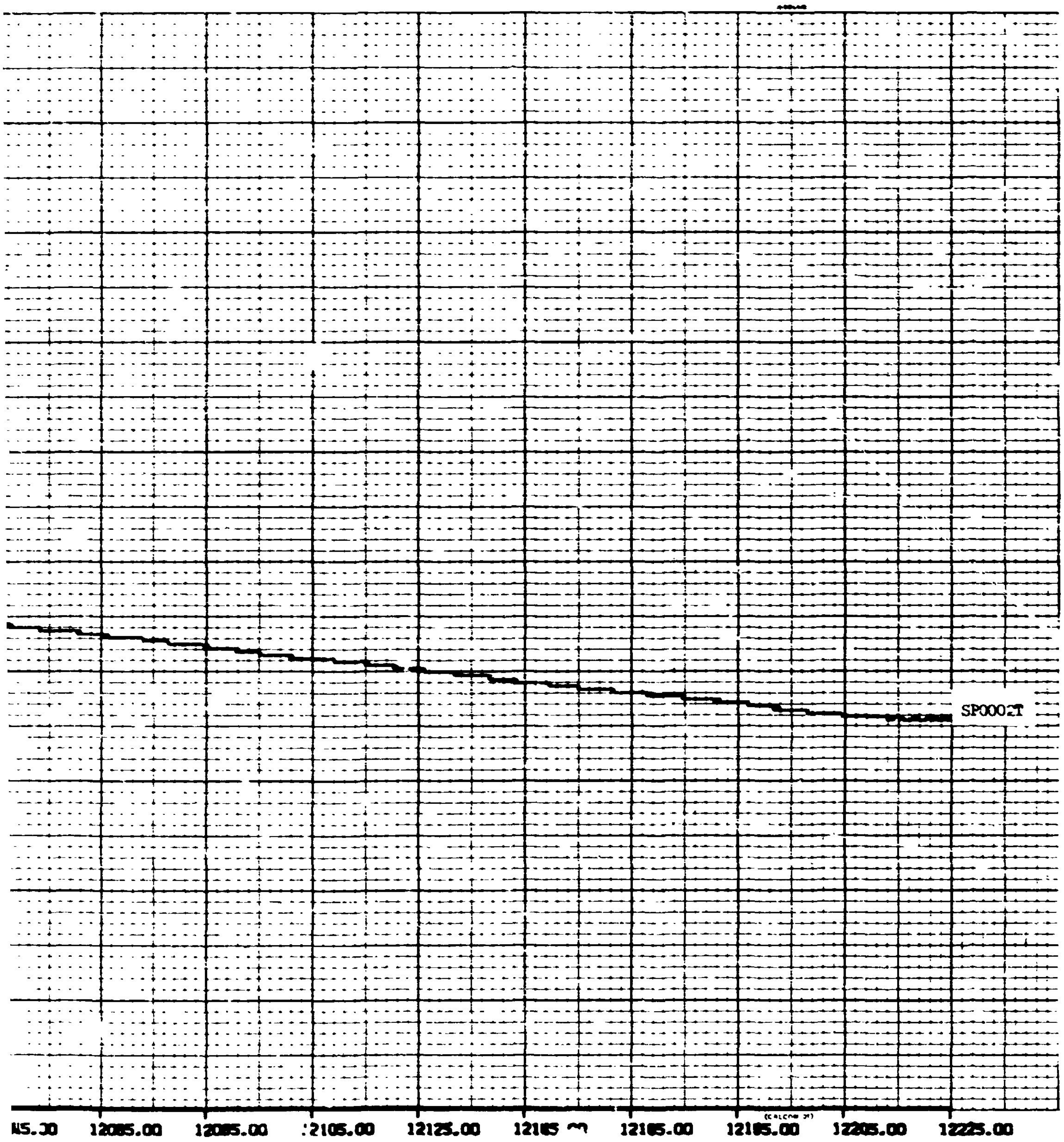

FOLDOUT FRAME 3

Figure 20.- Helium bottle temperature.

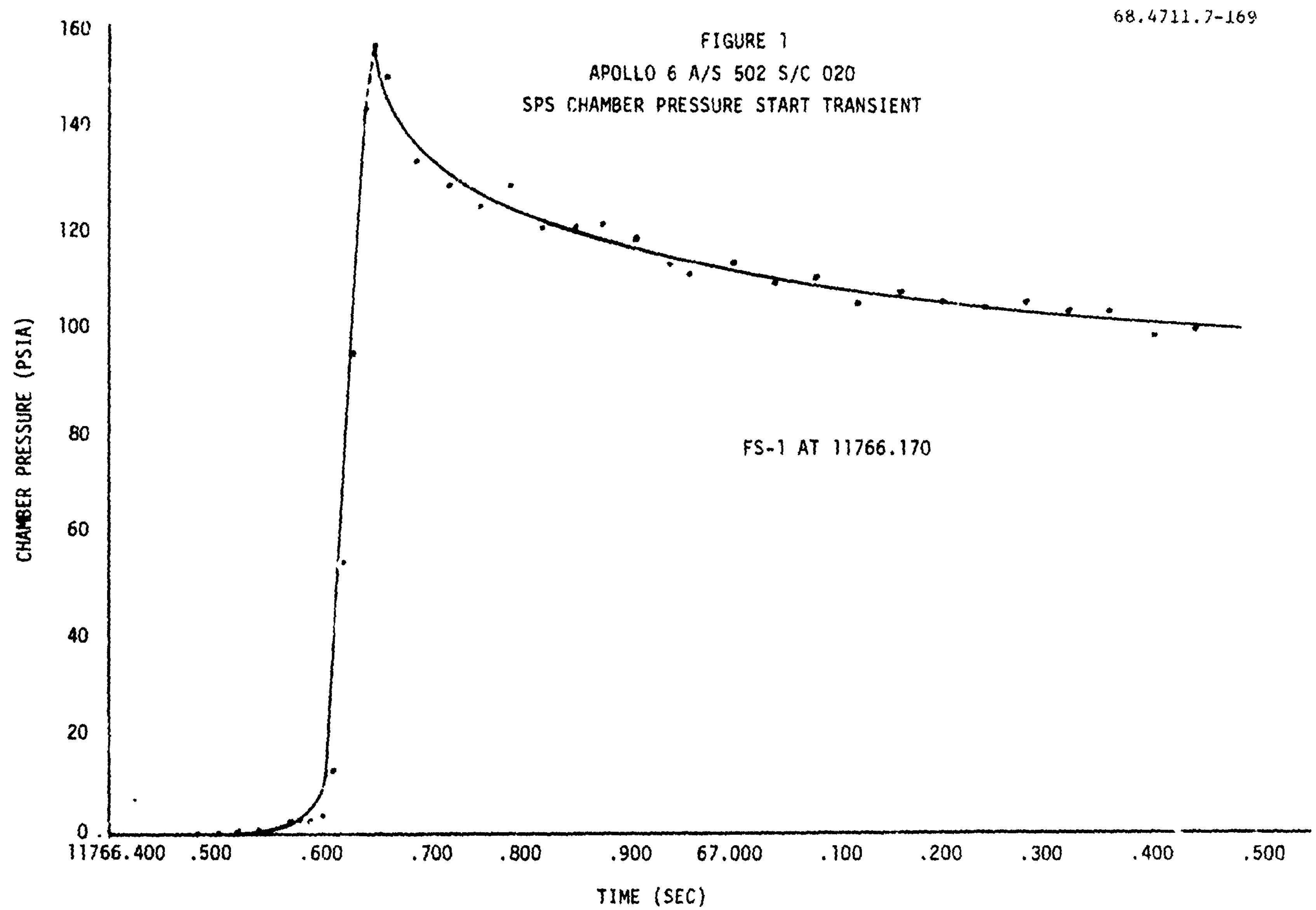

Figure 21.- Service propulsion system chamber pressure start transient.

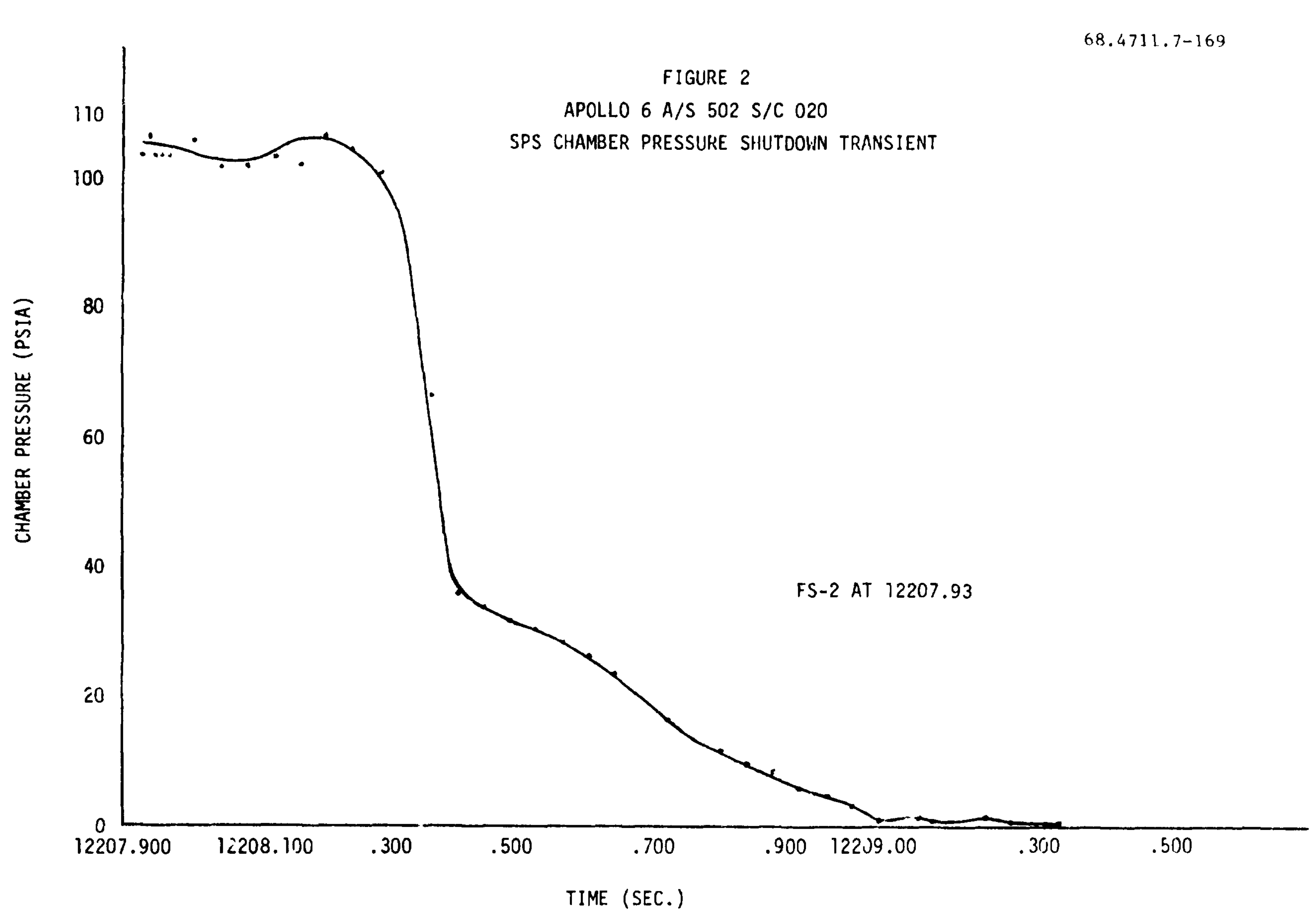

Figure 22.- Service propulsion system chamber pressure shutdown transient.

APPENDIX

PROPULSION AND POWER BINARY TAPE FORMAT

Record 1

Mission or test identifier information consisting of sixteen 36-bit binary coded decimal (BCD) words (96 characters).

Record 2

A BCD record containing measurement identification (ID) codes of eight alphanumeric characters each. Record length will depend upon the number of measurements requested but must be multiple of four words.

Record 3

Binary record as follows:

Word no.	Contents
1	Number of main frame pins on the commutator
2	Main frame delta time in milliseconds
3	Main frame pin number corresponding to the first measurement ID in record 2
4	Index value assigned to the first measurement ID in record 2
5	Main frame pin number corresponding to the second measurement ID in record 2
6	Index value assigned to the second measurement ID in record 2
.	.
.	.
.	

Word no.	Contents
n-1	Main frame pin number corresponding to the last measurement ID in record 2
n	Index value assigned to the last measurement ID in record 2

Record length will depend upon the number of measurements requested. All information in the above record will be fixed point.

Record 4 — Last

Binary records as follows:

Word no.	Contents
1	Index word (a) Bits 1 through 4 — Unused (b) Bits 5 through 12 — Index for data in word 2 (c) Bits 13 through 20 — Index for data in word 3 (d) Bits 21 through 28 — Index for data in word 4 (e) Bits 29 through 36 -- Index for data in word 5
2	Data
3	Data
4	Data
5	Data
6	Index word (same as described for word 1 except the indexes apply to data in words 7, 8, 9, and 10)

Word no.	Contents
7	Data
8	Data
9	Data
10	Data
.	.
.	.
.	.
n-4	Index word (same as described for words 1 and 7 except the indexes apply to words n-3, n-2, n-1, and n)
n-3	Data
n-2	Data
n	Data

In the records as described above, all index words will be fixed point and all data words will be floating point. Record length may vary from run to run but will be constant for any one run and must be a multiple of 60 words. Bit 1 of each index word is the most significant (leftmost) bit of the word. A time word equal to -20 000.0 indicates end of data on tape.